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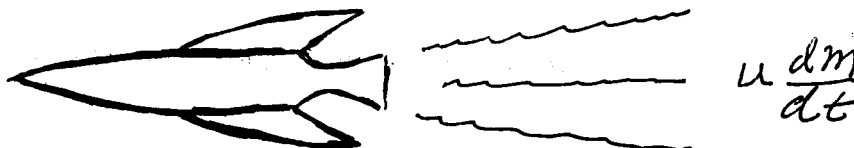
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## I. INTRODUCTORY PRINCIPLES

The subject "non-chemical propulsion" is so comprehensive that I cannot do justice to the subject in the time allowed. I nevertheless am going to use a portion of my time to discuss the theory of propulsion in the hopes that by so doing, you will obtain a better insight into the relative compromises associated with each of the propulsion systems.

First, consider a rocket in gravity free space. To obtain thrust, a



propellant must be ejected from the rocket. The reaction force on the rocket is calculated from Newton's laws as

$$F = \frac{d m u}{dt} = u \frac{dm}{dt} \quad (1)$$

where  $u$  is the constant ejection velocity of the propellant and  $\frac{dm}{dt}$  is the rate of propellant consumption. Now, let us assume that the propellant is ejected as a gas which has a total energy per unit weight of  $C_p T$ .  $C_p$  is the specific heat at constant pressure per unit weight and  $T$  represents the temperature of the gas before it passes through the exhaust nozzle. Now, as the gas passes through its exhaust nozzle,

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this internal energy will be converted to velocity. Conservation of energy at any point along the way then says that the thermal energy of the gaseous propellant plus the kinetic energy is equal to the original energy.

$$C_p t + \frac{1}{2} u^2 = C_p T \quad (2)$$

The maximum jet velocity for complete expansion is therefore

$$u = \sqrt{2 C_p T} = \sqrt{\frac{2 \gamma R T}{(\gamma - 1) W}} \quad (3)$$

where  $\gamma$  is the ratio of specific heats at constant pressure and constant volume and  $W$  is the molecular weight.  $R$  is the universal gas constant. Thus, from equations (3) and (1), we see that the thrust increases as the ratio of  $\sqrt{T/W}$  increases. We want high temperature, low molecular weight gases to give the best performance. Let me define the term "specific impulse" as the force in weight exerted when one weight unit per second of propellant flows. Hence:

$$I = \frac{(F/g)}{\frac{dm}{dt}} = \frac{u}{g} = \frac{1}{g} \sqrt{\frac{2 \gamma R T}{(\gamma - 1) W}} \quad (4)$$

jet velocity and specific impulse are essentially the same except for the gravitational constant "g".

From Newton's laws, the acceleration of the rocket in space is thus

$$F = \frac{m dv}{dt} = I g \frac{dm}{dt} \quad (5)$$

Integration gives:

$$\Delta v = I g \ln \frac{m(\text{initial})}{m(\text{final})} \quad (6)$$

Thus, the velocity increase of the rocket case and payload is proportional to the specific impulse and to the ln of the initial to final mass ratio for that stage. The velocity increment is independent of the size of the rocket in this relation.

A two-stage rocket will give twice the  $\Delta v$  of a one-stage rocket but the respective mass ratios are multiplied. For example, take rockets that have a mass fraction of 10/1 with a  $\Delta v$  of 10,000 feet/second per stage. The following table then pertains:

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<u>Number of Stages</u>	<u><math>\Delta v</math></u>	<u>Mass Fraction per Stage</u>	<u>Overall Mass Fraction</u>
1	10,000	10	10
2	20,000		100
3	30,000		1000

One might suppose from the above discussion that the highest jet speeds possible are always desired but this is not necessarily the case. While it is true that jet thrust increases with jet speed, the jet power increases as the square of the speed at constant propellant flow rate. Hence, jet power considerations may call for lower than maximum jet speeds. A qualitative rule states that the jet velocity should be comparable to the desired vehicle velocity. Thus, a jet airplane with a required speed of one or two thousand feet per second might require a jet exhaust speed of one or two thousand feet per second. Space travel would require very high jet speeds to arrive at the destination in a reasonable time period.

More quantitatively, the jet power neglecting the thermal energy is

$$P = \frac{1}{2} \dot{m} u^2 \quad (7)$$

From equation (1), the jet power per poundal of thrust is:

$$P/F = \frac{u}{2} = \frac{I_g}{2} \quad (8)$$

The propellant flow rate per poundal of thrust is:

$$\frac{\dot{m}}{F} = \frac{1}{u} = \frac{1}{I_g} \quad (9)$$

Hence, the product

$$\left( \frac{\dot{m}}{F} \right) \left( \frac{P}{F} \right) = \frac{1}{2} , \text{ a constant} \quad (10)$$

This relation is shown on Slide 1. Thus, if only low grade energy sources are available, such as in the chemical rocket, large mass flow rates will be required. The vehicle weight at take-off will be principally propellant.

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On the other hand, if high grade energy sources such as nuclear fission or fusion are available, then higher specific impulses (jet velocity) may be employed leading to lower fuel consumption rates. Under such circumstances, the power generation system weight may become as important or more important than the propellant weight. The trade-off between propellant weight and power plant weight gives a requirement for an optimum specific impulse.

Take electric propulsion, for example. The acceleration of a singly charged positive ion through a potential drop of only one volt would correspond to a chemical rocket combustion temperature of 11,600°K - and many thousands of volts acceleration are feasible. Hence, electric propulsion offers the choice of specific impulses covering the complete range from that of the chemical rocket up to that of the photon rocket with jet speeds approaching that of light. However, the power required increases with specific impulse and correspondingly the powerplant weight increases with specific impulse.

On the other hand, the total propellant load is proportional to the product of the propellant consumption rate and the time period that the propellant is being used. For a given mission, we may thus represent the propellant load by the rectangular hyperbola. Clearly, the sum of propellant plus power plant weight has a minimum suggesting an optimum specific impulse at the point of intersection of the straight line and the rectangular hyperbola. If the mission times are increased, higher required specific impulses result. Lower specific powerplant weights also suggest higher specific impulses.



## II. THE NUCLEAR ROCKET

We have seen the importance of specific impulse to the performance quality of rockets. The chemical rocket has a maximum specific impulse of perhaps less than 500 seconds. This limit results because the chemical reaction required to achieve the high exhaust temperature also gives a gas with compensatingly high molecular weight. On the other hand, a nuclear reactor might be used to heat hydrogen as the propellant to maintain a low molecular weight. In this case, the maximum temperature of the gas is limited by the materials available for constructing the reactor. Utilizing equation (4), the specific impulse then calculates to be in the range from 750 to 1,000 depending upon various assumptions. With such impulses, the nuclear rocket offers perhaps a 5 fold reduction in interplanetary spacecraft weight over that of the chemical rocket.

A schematic drawing of a solid core heat transfer type nuclear rocket is shown in Figure 2. The fission energy is liberated within solid materials of which the reactor core is composed. This heat is transferred to hydrogen as it passes through the axial heat transfer passages on the way toward the exhaust nozzle. The liquid hydrogen is pumped from the propellant tank through the nozzle walls and reflector for cooling purposes. The vaporized hydrogen then passes through the reactor and nozzle to produce thrust.

The heat of the reactor is, of course, derived from the fission of high atomic weight elements such as  $U_{235}$ . A neutron enters the Uranium atom to cause fission into several fragments of lower atomic weight. Additional neutrons are also released to cause other Uranium atoms to fission. The type of reactor design depends upon how these additional neutrons are conserved.

The neutrons are quite energetic when they are liberated from a fissioning atom. However, the probability of interaction with another Uranium atom is increased if the neutrons are slowed down. This can be accomplished by use of a moderating material in which the neutrons are randomized to thermal velocities.

A homogeneous thermal reactor is one in which the fissionable material is intimately mixed with the neutron moderating material as shown in Sketch 3(a). The heat deposited in this moderator by fission is removed by passage of hydrogen through an array of coolant tubes running from one end of the reactor to the other. In addition to slowing down the neutrons, the moderator must serve as a high temperature heat exchanger. Graphite and beryllium oxide are the only two materials that can reasonably serve this dual function. Graphite has the poorer moderating properties of the two, and therefore its use leads to larger core dimensions and

weight. On the other hand, beryllium oxide is limited to operating temperatures of at least 1000° F less than that of the graphite. Since specific impulse is so important, graphite is really the only contender in this homogeneous type reactor.

Unfortunately, hydrogen attacks hot graphite forming acetylene and other gaseous compounds. Hence, the use of graphite requires a protective coating on the heat transfer passages to prevent chemical reaction and corrosion in the hot hydrogen atmosphere. The Los Alamos Laboratory has spent a great deal of effort developing these coatings.

The first nuclear rocket reactors are graphite moderated. They are designed by the Los Alamos Laboratory and are undergoing tests at Jackass Flats, Nevada, in the KIWI program. Thus far, six KIWI reactor tests have been run at nuclear power. The last three of these were part of the KIWI-B program aimed at developing a basic core design that could be engineered for flight application. The last two power experiments were run using liquid hydrogen as the coolant with a regenerative, liquid-hydrogen-cooled jet nozzle.

Important results were obtained in these tests to indicate that the method of reflector drum control is an effective one and that the reactor can be started in a controlled manner with liquid hydrogen. However, reactor damage has been encountered. In the latest power test run on November 30 of last year, the KIWI-B4A reactor, which I might add is the favored basic design for flight development, encountered severe vibrations early in the test run. Examination of the reactor indicated cracking in almost all of the fuel elements and damage to certain thermal insulation components surrounding the core.

Non-nuclear simulation tests have since been run replacing the fuel elements with unloaded graphite. These tests were run with nitrogen, helium, and hydrogen gas flows with pressure drops through the core similar to those that exist during a normal reactor startup. Vibrations were encountered similar to those of the November nuclear test.

With hind sight, these non-nuclear tests should have been performed earlier. They were bypassed in the original program under the pressure to obtain full power tests under tight self-imposed time schedules in the hopes that the full reactor test would be successful.

The present program is being expanded to include all of the detailed, simulated, component and subassembly flow and mechanical tests that are required. In addition, full reactor tests under cold gas conditions and cold liquid and vibration tests will be included. When sufficient confidence exists that the reactor is well along toward a satisfactory design, then the major hardware development on the NERVA engine and the RIFT stage will proceed. Security prevents my being more explicit.

There are at least two other reactors that might be useful for nuclear propulsion. In the fast reactor shown on Figure 3(b), no moderator is employed. Hence, the uranium content must be increased to compensate for the poorer neutron cross section statistics for the fast neutrons.

Fast reactors can be made very small and can use the best available fuel-bearing or fuel-containing materials. The largest drawback stems from the fact that nuclearwise fast reactors are less efficient than moderated systems and a great deal more fissionable material is required for criticality. This leads to more difficult materials problems, since the volume of fissionable materials must be approximately equal to the total volume of all other materials in the core. Unfortunately, since fissionable compounds are not very satisfactory as structural or heat transfer materials, they must be contained within a refractory material such as tungsten, molybdenum, or the carbide of zirconium, hafnium or tantalum.

Development of fuel elements that contain 50 volume percent of fissionable material without penalizing high temperature performance is difficult. There are also difficult control problems associated with local thermal gradients during start-up. Nevertheless, the Argonne Laboratory is studying the feasibility of fast reactors with quite a bit of enthusiasm as a backup to the graphite reactor program.

At the Lewis Research Center, we have been more interested in a heterogeneous thermal reactor as a potential backup. In the heterogeneous arrangement (3c), the fuel elements are separated from the moderator. Hence the best materials for each can be employed. This type of reactor is not new. It received considerable attention by the General Electric Company in the ANP program in a reactor labeled HTRE 1. In that reactor, Slide 4, the nichrome-contained fuel element was inserted into insulated aluminum tubes surrounded by water. The air to be heated passed over the nichrome fuel elements mounted inside the aluminum tubes.

In the nuclear rocket application, tungsten would replace the nichrome in order to give higher gas temperatures.

The general arrangement of the fuel element matrix is shown on Slide 5. Concentric fueled plates are pictured in aluminum tubes that would be surrounded by the water moderator.

Slide 6 shows a sectioned view of a typical fuel element tube. The fuel element is formed of five concentric cylinders of clad tungsten- $\text{UO}_2$  material. The fuel cylinders are supported and spaced by fuel support pins. The upstream pins (to the left) pass through and are fastened to a tungsten fuel support tube. The fuel support tube runs the entire length of the reactor and provides a gap between it and the water-cooled aluminum tube which is filled with stagnant hydrogen. The stagnant hydrogen gap of about 1/8 inch thickness contains a thin molybdenum radiation shield. This technique of insulation reduces the heat loss from the fuel cylinders and hot hydrogen to a fraction of one percent of the full reactor power. Considering that 6 to 7 percent of the reactor power is deposited in the water anyway by neutron and gamma heating, this small additional heat load is hardly of any consequence.

Slide 7 shows a photograph of a full-scale model. An aluminum pressure vessel is completely filled with water except for the aluminum tubes which contain the tungsten fuel elements and flowing hydrogen. The water to hydrogen heat exchanger is divided into six equally spaced segments, one of which is shown at the top. The hydrogen from the nozzle cooling passages enters the tubes of this heat exchanger where it removes the heat deposited in the water. The hydrogen then enters the reactor inlet plenum. From this region, the hot hydrogen is expanded through the nozzle to produce thrust at a specific impulse of 800 to 900 seconds. The water moderator is circulated through the core and heat exchanger by means of a pump and inlet and outlet water plenum.

It is apparent that the high temperature problems of this reactor concept are concentrated within individual isolated small fuel elements. The remainder of the reactor is made entirely of aluminum which is water-cooled at all points. This major structural component can be developed to a high degree of perfection without resorting to full scale nuclear testing. In fact, one of the beauties of this whole reactor concept is that it is highly susceptible to component evaluation and improvement without requiring a full scale test in the early stages of development.

I should mention that natural tungsten has a high resonance capture for neutrons. The capture cross sections are shown on Slide 8. However, we want the neutrons to escape from the tungsten into the water to be thermalized. This desire can be accomplished by utilizing the tungsten 184 isotope rather than the natural metal.

The effect of enrichment of the tungsten 184 as well as the thickness of the water moderator on an idealized geometry is shown in Slide 9. The cell multiplication factor  $K$  is simply the number of neutrons produced by fission per neutron absorbed in the cell. It may be seen that  $K$  reaches a peak at 1.18 for natural tungsten but for enriched tungsten,  $K$  can be greater than 1.5. These peak values occur for water thickness between 0.5 and 1.0 inch.

Since the cell multiplication factor is a measure of reactivity for an infinite number of these cells, the excess above unity must be used for neutrons to be lost by leakage from a finite critical reactor size. It follows that a natural tungsten reactor would have to be very much larger than an enriched tungsten reactor.

The separation costs for tungsten 184 is quite a bit less than the cost of uranium. In this sense, construction of the reactor from tungsten 184 appears to be an economy.

In our preliminary experiments, we have been able to manufacture sample fuel elements that satisfactorily contained uranium in hot tungsten-clad geometries. The problem of insulating the aluminum tubes from the hot gas stream appears to have a simple solution. There are no serious chemical reactions between the hydrogen and the tungsten. Tungsten 184 can be produced in sufficient quantity without hampering the current uranium purification program. We are currently evaluating a proposed heat exchanger to cool liquid water with liquid hydrogen. And we are building hot hydrogen facilities to evaluate the nozzle heat transfer problems as well as to evaluate the non-nuclear integrity of the fuel element in a flowing hot hydrogen reactor simulated gas stream. Thus we have a good start on the preliminary research necessary to establish this nuclear reactor concept.

I would like to conclude this discussion of the nuclear rocket with a few comments concerning their use. To begin with, anything nuclear has political repercussions. While nuclear engines could feasibly be safely used to boost space payloads into orbit, the probability is high that such flights would not initially be permitted. Hence, getting to orbit will be accomplished by chemical means.

The graphite nuclear rocket is heavy even without nuclear shielding. Hence, nuclear propulsion could not be justified for the smaller missions. An approximate number to remember is that the vehicle take-off weight from a 300-mile parking orbit must exceed 50,000 pounds to justify nuclear propulsion over high energy chemical rockets. Of course, a manned round trip expedition to Mars via nuclear propulsion would require more than a million pounds of spacecraft in earth orbit.

On Slide 10, I have compared the performance of nuclear and chemical rockets. You can easily see that nuclear propulsion offers substantially higher  $\Delta v$ 's or substantially higher payload weights than can be used with high energy chemical rockets. This particular comparison used relatively low power nuclear rockets compared to current thinking. However, the advantages of nuclear rockets in large sizes is even more obvious.

For manned nuclear flight, careful consideration of the radiation hazard is of course required. In the early phases of the mission, the large propellant load can serve admirably for shielding. The shielding requirements for the terminal phases need more study but in general the feeling persists that the reactor shielding requirements are modest. People are much more worried about giant solar flares, Van Allen belts, and cosmic rays in about that order.

### III. THE SOLAR-HEATED HYDROGEN ROCKET

We have seen from equation (4) that the specific impulse of a rocket utilizing hydrogen as the sole propellant depends principally on the temperature to which hydrogen can be heated. In the nuclear heat transfer rocket, this limit is set by the properties of materials. Kraft-Ehrlicke and others have proposed that solar energy replace the nuclear reactor. In this way, the nuclear radiation hazards are avoided as well as the shielding problems. However, a new set of problems must be faced including those of the collector and the requirement for precise orientation relative to the sun. Also, because the sunlight only has about 1.34 kilowatts of energy per square meter at earth's distance, you can see that the required solar collector area could become very large. The nuclear rockets we discussed were on the order of thousands of megawatts. Also, the idea doesn't work in the shadow of a planet. I nevertheless wanted to call your attention to this idea.

### IV. ELECTRIC PROPULSION

The electric rocket refers to a rocket system requiring electric energy to accelerate the exhaust jet. Electric rockets can, in general, give exhaust velocities greater than are achievable by chemical means. However, the powerplant required to convert nuclear heat to electricity and the space radiator to dissipate the waste heat are heavy. Hence, accelerations of the spacecraft by electric propulsion will be very low - on the order of  $10^{-4}$  g. The electric rocket must thus be launched into orbit by some other propulsion system - probably the chemical rocket.

Once in orbit, the continuous application of thrust will add considerable energy to the space vehicle. The spiral path of a ship with a thrust of one pound for every 10,000 pounds of weight is shown in Slide 11. The moon's orbit is reached in 83 days. Escape from earth's gravitational energy occurs in 127 days. And with even this low but continuous thrust, faster trips could be made to the edge of the solar system with electric propulsion than with other propulsion means. The powerplant weight is heavy, of course. But the saving in fuel weight on extended space missions more than compensates this difficulty yielding higher payload fractions.

I will show later that electric propulsion allows high specific impulse. The primary reason for seeking high specific impulse is to reduce the jet fluid consumption and hence the required fuel load for a given space journey. The jet consumption rate is  $\dot{m} = F/I$ . (The thrust is now written in weight units). The fuel load for a propulsion time  $t$  is therefore

$$W_f = \frac{F}{I} t \quad (11)$$

On the other hand, the weight of the power generating equipment will increase with the required power output. The powerplant weight is

$$W_e = \frac{\alpha}{\eta} P = \frac{\alpha IF}{45.9 \eta} \quad (12)$$

where  $\alpha$  is in pounds per electric kilowatt and  $\eta$  is the efficiency of converting electrical energy to jet energy. The weight of the powerplant plus propellant is thus

$$W = W_f + W_e = \frac{F}{I} t + \frac{\alpha IF}{45.9 \eta} \quad (13)$$

These weights are shown schematically on Figure 1.

The minimum weight to thrust ratio is obtained when

$$I^2 = \frac{45.9 \eta t}{\alpha} \quad (14)$$

Thus propulsion time is always involved as well as specific powerplant weight when choosing the right specific impulse for a mission. Now the payload plus structural weight is equal to the total weight minus the sum of fuel plus powerplant weight

$$W_{p+s} = W_o - W_f - W_e \quad (15)$$

Inserting  $I$  from (14) into (13) and the resulting  $(W_f + W_e)$  into (15) gives

$$\frac{W_{p+s}}{W_o} = 1 - 2 \frac{F}{W_o} \sqrt{\frac{\alpha t}{45.9 \eta}} \quad (16)$$

Slide 12 shows the payload plus structural weight fraction that can be carried from a 300-mile orbit to a stationary orbit, to moon missions, and to Mars missions, all one-way trips. This figure was computed using optimum impulses for each journey and ratio of  $\alpha/\eta$  taking into account the loss in ship weight as the journey proceeds. Hence, the overall thrust to weight ratio varies somewhat along each curve. For the Mars journey, there is a coasting period following the thrusting portion of the flight. Clearly larger payload plus structural weight fractions can be carried for the longer



propulsion times on each mission. Also higher powerplant specific weights are allowed for a given payload fraction on long missions than for short. In general, the lower the specific weight, the better the performance. Much more sophisticated and more exact treatments of electric propulsion missions are of course in the literature.

As to electric propulsion hardware, I plan on dividing my discussions into two parts, one on accelerators and the second on power generation equipment.

Electric Propulsion Accelerators. Three types of electric rockets have been defined as shown on Slide 13. They are

(a) Ion accelerators: propulsion by means of charged particles that have been accelerated by electric fields.

(b) Plasma accelerators: propulsion by means of a plasma that has been accelerated by combination of electric and magnetic fields.

(c) Electrothermal accelerators: propulsion by means of a fluid that has been heated by electricity and is then expanded through a more or less conventional supersonic nozzle. I will discuss the electrothermal accelerators first.

The heating may be accomplished either in an electric arc or by means of a resistance heater. In either case, the performance is limited by the ability of materials to be cooled.

The arc heated rocket is shown schematically on Slide 14. The propellant, probably hydrogen, is used to regeneratively cool the nozzle and electrodes. The propellant then passes through the arc to the exhaust nozzle and to space. The arc is generally stabilized either by injecting the propellant into the heating chamber in a vortex flow or by the use of magnetic fields. A ballast resistor may be required between the generator and the arc chamber to promote further stabilization.

Experimentally, the arc-heated rockets are running at specific impulses below 2000 and at efficiencies of perhaps 60%. One hundred percent efficiency is of course out of the question because of the non-equilibrium lack of recombination of the dissociated propellant in the exhaust nozzle.

The resistojet is shown in Slide 15. In this arrangement, the propellant is energized by an electrically heated tungsten resistance. Hence, a maximum specific impulse of perhaps 1000 is feasible. The specific impulse on electrothermal rockets roughly follows equation (4).

The electrothermal rockets are useful for attitude control and for near-earth missions where the specific impulses of the electrothermal rocket are near optimum. The electrothermal rocket can also be justified for missions where the electric powerplant is already on board the spacecraft for other means. In this case, the propulsion system would not be charged with the weight of the power supply. (A communication satellite, for example). However, the electrothermal rocket is not competitive with the nuclear rocket or with ion propulsion, for example, on deep space missions.

This latter conclusion follows from a consideration of the powerplant. You see nuclear reactor heat is converted to electricity by direct or indirect means at no more than 20% efficiency. Hence, at least four times the required power must be radiated to space in a radiator that is both cumbersome and heavy. It does not make sense to convert nuclear heat to electricity in order to produce heat at such a weight penalty. Hence, electric propulsion is in general not competitive at specific impulses less than about 2000 with the simpler open cycle nuclear rocket. This is just the specific impulse range of the electrothermal devices.

Ion accelerators: An ion rocket, Slide 13, contains a source of positive ions, a set of electrodes for accelerating the ions, and an electron source for neutralizing the beam following the acceleration. The beam leaving the engine must, of course, be electrically neutral if thrust is to be maintained.

The jet velocity of the ion source is related directly to the voltage drop across the accelerator portion of the engine. If the gain in kinetic energy of a charged particle  $\frac{1}{2} m u^2$  is equated to change in potential energy  $ne\phi$  with some juggling of units, we obtain the specific impulse as

$$I = 1.42 \times 10^3 \sqrt{\frac{ne\phi}{W}} \quad (17)$$

where  $\phi$  is the potential drop across the accelerator in volts and  $W$  is the molecular weight of the propellant.  $n$  is the number of electron charges on the particle. By comparing this relation with equation (4) obtained earlier, you can easily see that temperature and voltage are interchangeable. For singly charged hydrogen molecules, one volt of acceleration would give a specific impulse of about one-thousand. Thus, a great range of specific impulses are possible simply by adjusting the potential drop across the accelerator.

On the other hand, there are fundamental limits to the beam currents of an ion accelerator. If too many charges of like sign try to pass simultaneously through the accelerator, then the electric field due to the ion cloud is opposed to field due to the accelerator potential. Thus, the intensity of an ion beam is generally limited by space charge considerations in accordance with the Langmuir-Childs relation:

$$\dot{z} = 5.56 \times 10^{-12} \sqrt{\frac{\epsilon}{\mu}} \frac{\phi^{3/2}}{L^2} \quad (18)$$

where  $\epsilon/\mu$  is the charge to mass ratio in coulombs per kilogram,  $\phi$  is the potential in volts, and  $L$  is the distance from the ion source to the accelerator grid.

There is, of course, the possibility of using higher acceleration voltages than required for the optimum impulse, followed by a deceleration electrode to bring the impulse back down to the required value. This accel-decel trick gives an improvement in beam current density over the conventional limit set by equation (18).

If accel-decel is not employed, the thrust per unit area may be obtained by Newton's laws combined with equation (18):

$$F/A = 8 \times 10^{-13} \left(\frac{\phi}{L}\right)^2 \approx 1.8 \times 10^{-9} \left(\frac{\mu}{\epsilon}\right)^2 \frac{I^4}{L^2} \quad (19)$$

Thus if the specific impulse is fixed by the mission, the thrust per unit area increases with the square of the mass to charge ratio. Hence, the desire for high molecular weight ions. Thus, the thrust density is set by the mass to charge ratio while the specific impulse is adjusted by means of the accelerator voltage within voltage breakdown limits.

Following release of the positive ions at optimum impulse and ground potential, electrons should be blended with the beam as soon as possible to neutralize the charge. The electron velocity should equal the ion velocity. The acceleration voltage of the electrons should therefore be smaller in proportion to the mass ratio than that of the ions. The mass ratio of an electron to a cesium atom is  $4.12 \times 10^{-6}$ . Hence, a 10,000 electron volt cesium ion would have the same speed as a 0.04 volt electron. This low energy represents an electron at a temperature of only 40°F. Hence, all that is required for beam neutralization is to place a hot electron emitting filament in the beam downstream of the accelerator. With almost insignificant loss in the positive ion energy, the electrons are simply dragged along by the positive charges to produce beam neutralization.

Slide 16 shows a schematic diagram of a cesium ion engine. Cesium gas is first produced in a vaporizing chamber. It is then brought into contact with hot tungsten surfaces. In early experiments, the tungsten surfaces were composed simply of a labyrinth of tungsten strips. In most of the modern engines, porous tungsten is used through which the cesium gas passes.

The ionization potential of cesium is 3.87 volts. This represents the work required to remove an electron from the cesium atom. On the other hand, the work function or the containment energy of electrons in tungsten is higher than 3.87 volts. Hence, each atom of cesium is ionized as it bounces off a tungsten surface. The tungsten must be kept hot simply to keep the surface clean and free from cesium condensation. A layer of cesium only a few molecules thick would stop the ionization process.

The ions are then accelerated and focused in the accelerator portion of the engine. Acceleration and deceleration electrodes might be employed with the last electrode at essentially space (or ground) potential. Finally, an electron emitting filament is included to neutralize the ion beam.

Reasonably high vacuums must be used in the evaluation of ion engines. Even at  $10^{-5}$  Torr. there is a sufficient supply of neutrals so that charge exchange can occur in the beam between neutrals and ions. This leads to beam defocusing and electrode erosion due to sputtering. As a matter of fact, the beam will neutralize itself without the aid of the electron emitting filament at these pressures. Hence, facility pressures should be on the order of  $10^{-6}$  Torr. or better.

Slide 17 shows an ion beam in one of our Lewis Research Center tanks. This photo is slightly faked in that under high vacuum conditions, the beam is invisible. For this picture, we raised the tank pressure enough so that the ion bombardment of the neutral background gas produces the glow.

The electron bombardment ion engine invented by Harold Kaufman at the Lewis Research Center is shown on Slide 18. In this engine, the ionization is produced by electron bombardment. The electrons are emitted from a filament along the axis and are attracted to the outer concentric cylindrical shell by a positive voltage of, say, 100 volts. An imposed axial magnetic field, however, forces the electrons to gyrate around the field lines in epicycloidal paths around the axis. Mercury ions are produced by electron bombardment in this annular ionization region. They drift from here through the discharge ports to be accelerated toward space. Successful electron bombardment engines have been produced using permanent magnets in order to minimize the power losses. They have also been run successfully using either mercury or cesium as the propellant.

The efficiency of several thruster types are compared in Slide 19. Generally speaking, thruster efficiency improves with increasing specific impulse simply because the beam energy is increasing while the losses remain unaltered. On the contact ionization engine, the principal loss is associated with the requirement that the tungsten ionizer be kept hot. This engine is thus severely penalized for specific impulses below five or six thousand but performs quite well at values on the order of 20,000 seconds or higher. For this engine, the ionization mass fraction is essentially 100%.

On the other hand, the ionization mass fraction of the electron bombardment engine is only about 80% while the inherent losses are relatively low. This engine is thus more suitable at the lower specific impulses but might lose out at the high values.

I also show on this chart an estimated performance curve for colloidal particle accelerators. Experiments on these engines are so preliminary that we really don't yet know what performance can actually be obtained. You will recall from equation (19) that the thrust per unit area is proportional to the square of the mass to charge ratio ( $\mu/e$ ). Also, if the mass to charge ratio is very high, then the acceleration voltage will be very high. The idea is to produce colloidal particles of fractional micron dimensions, spray charges on them, and accelerate them to space taking care to neutralize the beam. If the charged mass fraction can be upped to nearly 100% and if the particles all have essentially the same mass to charge ratio, then the kind of performance shown can probably be achieved. Time and more research will tell.

At Lewis, we are producing the colloidal particles in a condensation shock generated by flowing vaporized aluminum chloride through a supersonic nozzle (Slide 20). Charges are then sprayed on and the colloidal particles thus generated either by corona discharge or by electron bombardment. The actual hardware for these two arrangements is shown in Slides 21 and 22. I won't say much more except that the preliminary results we have obtained are encouraging for low specific impulse engines. Some other laboratories are studying the charging of aerosol sprays to accomplish the same desired performance.

Another potential ion source for electric propulsion (Slide 23) is due to Von Ardenne. This source utilizes an electric arc to ionize the jet. The arc electrons proceeding forward from the filament are reflected by the positive ion accelerating field. They are also confined to a region near the axis by an intense magnetic field in a Phillips ion gage arrangement. Thus, the electron bombardment of the propellant in the arc gives nearly 100% ionization and perhaps an order of magnitude larger current densities than would be predicted from the Langmuir-Childs space charge limit equation. This latter effect is due to the fact that the ion beam is essentially neutralized in the low voltage or most critical portion of the ion acceleration history.

Still another potential source is the so-called Hall current accelerator (Slide 24). In this source as yet very much in the research stages, ionization is produced by electron bombardment in a region containing an axial electric field and a radial magnetic field. The ion cyclotron radius of the electrons is small compared to the apparatus dimensions so that the electrons are confined to gyrate on cycloidal type paths on a cylindrical surface about the axis. On the other hand, the ion cyclotron radius is chosen as large compared to the apparatus dimensions. In this manner, the ions leave the magnetic field before they have an opportunity to complete their cycloidal paths. Like the Von Ardenne arrangement, this source also has space charge neutralization in the low velocity regions of acceleration. Hence, the current density may be much higher than predicted by the Langmuir-Childs space charge limit (Equation (8)).

Plasma accelerators: Plasma accelerators are the third type listed on Slide 13. A plasma is an ionized gas containing equal numbers of positive and negative charges - and hence is electrically neutral. This plasma can serve as a conductor and hence can be accelerated by electromagnetic forces.

Now both positive and negative charges will spiral around magnetic field lines as shown on Slide 25. If an electric field is superimposed, the charges will alternately accelerate and decelerate with corresponding changes in path curvature. As a result, both positive and negative charges move through space in the same direction with an average drift velocity:

$$\bar{u} = \frac{\bar{E} \times \bar{H}}{H^2} \quad (20)$$

where  $\bar{E}$  and  $\bar{H}$  are the electric and magnetic field vectors respectively. This drift velocity corresponds to the zero power or idling speed of a motor. If energy is to be added to the plasma, the electric field vector must be tilted toward the direction of the desired acceleration. Accelerators (Slide 26) utilizing these crossed electric and magnetic fields are called  $\bar{E} \times \bar{H}$  accelerators. Space charge neutralization is, of course, unnecessary in plasma accelerators since both positive and negative charges move in the same direction.

A second type plasma motor may be visualized by having the plasma serve as a conductor in a magnetic field. The force on the plasma is then

$$\bar{F} = \bar{J} \times \bar{H} \quad (21)$$

where  $J$  is the current. Now a current can be generated by positive charges moving in one direction and negative charges moving in the opposite sense. Hence, here again, both positive and negative charges on the average move together so that space charge neutralization is unnecessary.

A simple form of a plasma motor is shown on Slide 27. An arc is struck between two parallel conductors. The resulting current generates a magnetic field that propels the plasma arc along the rails. The favored geometry for rail accelerators utilizes coaxial conductors with electric current flowing radially through the plasma.

The magnetic pinch effect may also be used to obtain magnetic plasma projection, one form of which is shown on Slide 28. An arc is struck as a continuation of the center conductor of a coaxial cable. The pressure associated with the confining magnetic field of the current carrying plasma plows the plasma toward the axis during the condenser discharge to give extreme pressure and temperature. The plasma squirts out through the hole on the axis.

A magnetic mirror may also be used for trapping and projecting plasma. In a magnetic mirror (Slide 29) the plasma is trapped in the magnetic trough between regions of higher field strength. If the "mirror" is then shifted along the axis at a continuously increasing speed to the desired plasma projection velocity, the plasma will also be accelerated accordingly as a surf board rides the waves.

You can see that there are many ways to accelerate plasma - almost any form of linear motor where plasma replaces the conductors will serve. I haven't even mentioned the AC self-induction schemes.

Most of the plasma propulsion ideas are not yet competitive with ion rockets. The efficiency is either too low or the hardware is too heavy. Nevertheless many experts believe that the day will come when ion sources are obsolete. Combined ion and plasma sources such as the Hall current accelerator may well be better than either alone. In any case, a great deal of progress must be made before the propulsion world will get excited about plasma sources. On the other hand, electric propulsion looks most promising for missions requiring high specific impulse. For such missions, two sources are almost to the useable stage of development now - the cesium contact ionization source and the Kaufman electron bombardment engine. The power production system is much more likely to delay the practical use of electric propulsion than would the accelerator.

#### Power Generation Systems:

The two keynote problems of the power generation system at present are (1) to achieve a sufficiently light powerplant and (2) to achieve sufficient endurance. An indication of these problems is shown on Slide 30 where electric propulsion is compared with the nuclear rocket for a manned Mars mission. Clearly, the powerplant specific weights should be less than about 20#/kw and reliability should be proven for times on the order of 500 or 600 days. The intersection of the nuclear rocket curve with those of powerplant specific weight generates in itself a curve similar to that shown on Slide 31. These two figures utilized different mission assumptions so there is only qualitative agreement between them.

There are presented here (Slide 31) several missions. On each of them, the powerplant specific weight must be lower for a given trip time than the presented curve if electric propulsion is to exceed the capabilities of the nuclear rocket. Even though there is no universal curve, it is clear that longer trip times permit higher specific powerplant weights. However, the reliability requirements become more extreme. Now, one can imagine that reliability could improve if powerplant weight is allowed to increase.



One could then superimpose a curve of endurance time on this one as a function of powerplant weight. The optimism for electric propulsion then is described by these limiting curves. If the powerplant specific weight is too high or the endurance is too poor, then electric propulsion would lose its competitive status.

Let us dwell for a moment on this question of endurance. One year has about 8800 hours so we are asking for at least 10,000 hours of trouble-free operation. An automobile would travel 300,000 miles at 30 miles an hour in that time. Surely some trouble would be expected. So our space powerplant must be much better than our automobile. No one knows how reliable a space powerplant can be, but the longest running time so far has been a factor of about 20 too low. In other words, quite a bit of optimism is required to suppose that the reliability goals can be achieved in the next few years. Also we must consider the other problem simultaneously - that of obtaining low specific weights.

The heaviest component of an electrical power generating system is the radiator for eliminating the waste heat. It is also the most vulnerable to meteoroid damage. The radiator for a 10 megawatt system might weigh four or five pounds per kilowatt if it did not have meteoroid protection. With protection, the weight could easily be 20 or 30 lbs. per kilowatt for the long time missions. Because the area of the radiator is strongly temperature dependent, there is a strong temptation to run the system at the highest possible temperature to reduce radiator area and hence, system weight. The limiting temperature is set by material corrosion difficulties in proposed Rankine cycle liquid metal systems diagrammed in Slide 32. High temperatures imply refractory metal loops that must be developed in an oxygen-free environment. So the problems are not easy.

The possibilities of using light weight material might raise the question of how high one should increase the radiator temperature. A beryllium radiator, for example, could feasibly be operated at 1400°F. Beryllium is one-fourth as heavy as conventional high-temperature materials. Thus a "conventional" high-temperature radiator would have to operate quite a bit hotter to break even with beryllium on a weight basis. Beryllium, on the other hand, may have unacceptable fabrication problems. Or perhaps the radiator tubes might shatter under meteoroid impact, or launch vibration conditions. So the beryllium radiator is still speculative.

The rest of the system includes a pump which boosts the liquid metal through a heat exchanger where it is vaporized. This vapor is then passed through the turbine and hence to the radiator where condensation takes place. Hence, the radiator temperature is reasonably constant throughout because of the two phase condensing process. Reactor heat is supplied to the heat exchanger by means of a second pumped liquid metal loop.

The Rankine cycle system appears to offer promise of a light weight system for the high power levels required for electric propulsion. However, two comments should be made. First of all, there are some real tough engineering problems. These include, in addition to all of the radiator and meteoroid damage problems, all those connected with liquid metal erosion and corrosion problems, with sludging and radiator clogging associated with material transfers; the difficulties associated with obtaining reliable turbine materials, and bearings lubricated by liquid metals, and seals; with condensation problems in the turbine, and the erosion and performance degradation associated with turbine exhaust moisture content, with condensation and fluid distribution problems in the radiator under zero "g" conditions, and with probable restart difficulties associated with freezing and sludging in the tubes. Some of these would necessarily have to be evaluated via costly space experiments. And I have not mentioned any of the problems associated with the reactor or with the difficulties of launching cumbersome radiators and of maintaining fluid system integrity against leakage of the liquid metals to space. With all of these difficulties, the Rankine cycle power plant development will be neither easy, nor quick and it likely will be very expensive.

Secondly, with so many difficulties and unknowns, achievement of the required reliability will be a long time in coming.

Rankine cycle steam power plants have also been proposed by the Astra Corporation. They look promising at the moment, but the studies are very preliminary.

Many of the listed difficult engineering and materials problems associated with two phase liquid metal systems can be avoided by utilizing the all gas Brayton cycle diagrammed in Slide 33. Using an inert gas such as neon or argon, most of the corrosion problems vanish. Hence, higher temperatures can perhaps be utilized in the cycle. The unit could be canned, thus eliminating the problems of seals on the alternator. The use of gas bearings might lead to a system with almost indefinitely long time reliability, and shut down and restart should be easier than on a Rankine cycle powerplant. What's more, we have a wealth of technical and engineering experience on Brayton cycle machinery from the turbojet and turboprop engine studies.

On the other hand, the radiator on the Brayton cycle is bulky. There must be a large temperature drop across the radiator to keep the machine running. Because radiation to space follows according to  $T^4$ , the low temperature portions of the radiator are very much less efficient than the high temperature portions. Hence, to really capitalize on a Brayton cycle system for electric propulsion would require operation at much higher temperature levels than we are accustomed to considering in order to keep

the specific weight down to useable values. With inert fluids such as Neon, higher temperatures can certainly be visualized. However, this means a new reactor development that might also be beyond our current technology.

Creep and stress rupture considerations on materials suggest a top reactor fuel element temperature of 2800°F. to 3000°F. for 10,000 hours of life. This would probably be a fast reactor. Perhaps turbine inlet temperatures of as high as 2500°F. could then be considered. Such a system would have specific weights sufficiently low to be attractive for electric propulsion - on the order of 15 pounds per kilowatt. At 2040°F. inlet to the turbine, the specific weight is about 25 #/kw.

Some discussion should be included on the status of thermionic-converter space power systems. The thermionic converter boils off electrons from the emitter, which then progress to the collector (Slide 34). In this manner, heat is directly converted to electricity by differences in temperature and work function between the emitter and the collector.

The power level of the vacuum thermionic converter is, of course, space-charge limited. Therefore, if reasonable spacing between cathode and anode are employed, an easily ionized gas such as cesium must be inserted to neutralize the electronic charge. The resulting "plasma thermionic converter" has received considerable interest as a potential source of space power.

The theoretical Carnot efficiency of the plasma thermionic converter ranges from 25 to 50 percent. Experimental efficiencies have been about one-third of these values, or a maximum of about 17 percent. The remaining heat energy must be discharged to space by means of a radiator. Unfortunately, the higher efficiency occurs with the lowest anode temperature which suggests a larger radiator. When the system weight including the radiator is minimized, the efficiency is approximately 10 percent, or perhaps a little higher with the new meteoroid data.

When we decide to use the thermionic converter in a space power system, we must decide whether to install the elements in pile or in an out-of-pile arrangement. The out-of-pile design is much easier and straightforward. A liquid metal or gas loop would carry reactor heat to the individual cathodes. However, the limiting temperature of the liquid metal system occurs in the reactor with the cathode at a still lower temperature. In this arrangement, there is perhaps a 600°F. penalty on the maximum cathode temperature leading to estimated system weights so large that we may draw the conclusion that out-of-pile thermionic conversion systems are not interesting for electric propulsion at this time. The conclusion depends strongly on the maximum feasible temperature of the system. The higher the temperature, the more feasible the out-of-pile arrangement.

A schematic diagram of an in-pile thermionic converter system is shown on Slide 35. In this configuration, the cathode on each thermionic unit is fueled with uranium. The reactor then consists of an array of thermionic elements arranged into a critical assembly of fueled hot cathodes. The anodes must, of course, be cooled. Thus, the reactor and the power generation equipment are combined into one unit.

Perhaps ninety percent of the energy so generated must be carried to the space radiator by means of a working fluid. Because the converter is a high-temperature device, the anode cooling and the transfer of heat to the radiator are accomplished by a liquid-metal system. Hence, the thermionic converter has the same limitations on performance due to the use of liquid metals as the Rankine cycle rotating-machinery device. The thermionic converter may operate at higher temperatures - turbine inlet temperature corresponds to anode temperature - but gains from this difference are offset at present by the lower efficiencies of the minimum-weight diode system.

Studies have been conducted on the use of gaseous cooling of the anode to raise the operating temperatures. In these studies, the pumping power to circulate the cooling fluid was unreasonably large except when large temperature drops across the radiator were employed. Then the radiator became both large and, with meteoroid protection, too heavy. Hence, gas-cooled thermionic conversion systems currently are not interesting for electric propulsion.

For that matter, no one has yet designed a satisfactory liquid-cooled thermionic power system for space. One might propose a reactor composed of a critical assembly of thermionic diodes, each with its uranium-fueled cathode. The engineering problems associated with balancing the nuclear characteristics of such a reactor with the diode thermal and electrical requirements, including the multiplicity of series and parallel groups of diodes, each cooled with a properly insulated liquid-metal system connected to a common radiator, is challenging to say the least. Add to this, the requirement for replaceability of each radioactive diode unit upon failure and the problem becomes even more difficult. When engineers are actually faced with this design job, they may find that the optimistically low estimates of the weights of the thermionic conversion systems sometimes included in the literature will grow to equal or surpass the weight estimates of more conventional approaches.

There is still another major problem with the thermionic system. It requires a relatively heavy power conditioning system to provide the proper voltage and currents for electric propulsion. When I consider the horrendous difficulties in arriving at a satisfactory engineering design for the thermionic system, I come to the conclusion that the thermionic system has to show a lot more progress before it can compete with dynamic power systems for electric propulsion. Its development is probably more difficult than the high temperature liquid metal Rankine cycle system.

You can see from this less than optimistic discussion that space power in sizes required for electric propulsion will not come easy. The systems that will provide this power with low enough specific weights and high enough reliability for man-rated interplanetary flights are a long way in the future. Even the basic research so necessary prior to a development phase is moving slowly and with great difficulties and expense. One could easily conclude that manned planetary flights using nuclear rockets are more likely to be undertaken first. Of course, electric propulsion may be used earlier in much less important applications such as guidance and control, satellite orientation, and satellite orbital adjustments.

## V. GASEOUS CORE NUCLEAR ROCKETS

The goal of the gaseous core nuclear rocket is to produce specific impulses above 1000 with a thrust to weight ratio on the order of unity or larger. Using hydrogen as the propellant, the exhaust jet must therefore be considerably hotter than the melting point of known materials. The trick is to heat hydrogen in a gaseous uranium reactor without losing too much uranium.

Cost can be used as a measure of the required hydrogen-to-uranium flow ratio. If it costs \$200 per pound to place hydrogen in orbit and \$7000 per pound to produce uranium, this hydrogen-to-uranium weight ratio is 35 to 1.

A typical gaseous core reactor might be 10 feet in diameter and 10 feet long. A uranium partial pressure of about 25 pounds per square inch is required to maintain nuclear criticality. The hydrogen pressure should be as high as possible within the limits of reasonable practice.

Let us suppose that 2000 pounds per square inch is reasonable. Then if hydrogen and uranium flowed through the reactor together, the hydrogen to uranium mass flow ratio would be

$$\frac{2000}{25} \times \frac{2}{235} = .68$$

which is far below the 35 to 1 required. Hence, we must increase the residence time of the uranium relative to the hydrogen by a factor of about  $35/.68 = 51.5$ . The struggle to find a good gaseous core reactor concept revolves about this problem.

Let us assume for the moment that we have a good cavity reactor concept. The cavity will surely be surrounded by a thick moderator and neutron reflector as shown on Slide 36. Heat will be generated within this moderator due to absorption of neutron and gamma radiation. This heating amounts to about 10% of the reactor power and must be removed regeneratively by the flowing hydrogen.

The maximum temperature of the moderator might be 5000°R. Hydrogen at 5000°R would have a specific impulse of about 900 seconds. Since this represents 10% of the heat, the jet specific impulse can be perhaps 900 multiplied by  $\sqrt{10}$ . Thus, a specific impulse of about 3000 seconds becomes an upper limit for the performance of gaseous core nuclear rockets.

The pressure shell to contain the required reactor pressures of 1000 to 10,000 pounds per square inch is sufficiently thick so that no reactor shield is required. Nevertheless, the total weight of the reactor including moderator and pressure shell is from 250,000 to 500,000 pounds. Hence, in a mission comparison with more conventional nuclear rockets, the gaseous-core rocket system would likely require a fuel load of more than 500,000 pounds to capitalize on its higher specific impulse. Thus more than 1,000,000 pounds in orbit would be required just for the engine and fuel load. To this would have to be added the payload and structural weight requirements. You may thus get some feel for the size of the mission before gaseous core nuclear rockets can be justified.

An early suggestion for a gaseous core reactor is shown in Slide 36. Tangentially entering hydrogen passes radially inward through a gaseous uranium vortex. Hopefully, the centrifugal forces associated with the heavier uranium molecules would be balanced by the diffusion drag of the inwardly moving hydrogen. The hydrogen would ultimately move along the axis to the exhaust nozzle as shown in Slide 37.

Unfortunately, the drag produced by the flowing hydrogen is so great that excessive loss of uranium will occur unless the hydrogen flow rates are limited to very low values. Hence, in a single-tube vortex reactor, only low thrusts could be obtained without excessive loss of uranium.

One way to avoid this difficulty is to use multiple vortex arrangements as are shown in Slide 38. Criticality is achieved by the combination of many gaseous uranium cores. These may either be materially separated, as in the upper left diagram, or established by a matrix injection pattern, as shown in the square box drawing. These schemes were proposed by Jet Propulsion Laboratory and Space Technology Laboratory. Both have a major problem of cooling the enclosed hardware.

Instead of passing all of the hydrogen through the uranium vortex to the core as on Slide 36, an alternative arrangement is to bypass part of the hydrogen to flow axially outside the uranium cloud to an annular discharge part. The United Aircraft Corporation is studying this arrangement.

The Lewis Research Center's coaxial jet reactor is illustrated on Slide 39. The central core of uranium gas would be injected at a much slower speed than the coaxially moving hydrogen. Hopefully, the mixing processes can be tailored to minimize the uranium loss rate. A hydrogen buffer layer would be added with an intermediate velocity profile between the uranium and the outer hydrogen layer to serve this purpose. The hydrogen to uranium velocity ratio should be 50 to 100 or higher for reasonable fuel conservation.

On the other hand, the velocity difference between the hydrogen and uranium layers can be eliminated entirely by using tangential entry and exit of the fluid as shown on Slide 40. The cylindrical uranium core is injected through the two end walls with an angular velocity of rotation to match that of the hydrogen buffer layer. The main hydrogen propellant enters and leaves the reactor tangentially. Small quantities of axially flowing hydrogen can be injected in the end walls for cooling purposes and to match the uranium axial velocity component generated by uranium replenishment. The end walls can even be rotated to eliminate the usual secondary flows.

In all of these reactors, the principal heat transfer mechanism to the hydrogen is by radiation, and the hydrogen must be continuously seeded with graphite powders and other materials to absorb the radiant heat before it reaches the containing walls. This is only one of the many very difficult research problems the cavity reactor faces.

In fact, it is real tough to plan meaningful, definitive experiments to evaluate cavity reactor concepts on small scale. There are grave nuclear and fluid mechanic stability problems that might require simulation of the reactor at close to full scale and full power operating conditions. Thus, the research decision to evaluate the feasibility of a cavity reactor concept may require a hazardous multi-billion dollar program.

## VI. PROJECT ORION

You have all probably had the joy of propelling a tin can into the air by means of an exploding firecracker. A rocket could conceivably be designed to fly by means of a succession of carefully timed firecrackers exploding in the vicinity of that tin can. The ORION concept is similar except that a succession of small nuclear explosions replace the firecracker - and a large spaceship resembling a city water tower in size replaces the tin can. Because the ship must withstand the agonies of nuclear explosions, naturally heavy shipbuilding construction and assembly methods are required. The design must include techniques for minimizing the destructive effects of the strong shock load, high temperature radiation, and other hazards associated with nuclear explosions in the near vicinity of the ship. And if men are on board, they must somehow be isolated from the large periodic accelerations that could result. You can see that such a spacecraft would be heavy - with weights comparable to that of a gaseous core nuclear rocket. Its use would therefore be restricted to large payloads involving requirements for substantial velocity increments.



To make the basic idea appear plausible, imagine that a nuclear bomb explosion converts the bomb material to an expanding gas at, say, 20,000,000°F. In a vacuum, this gas which is assumed to be expanding uniformly in all directions could reach an ultimate radial speed from Equation (3) of say, 1.8 million feet per second. If a plate of heavy material intercepts and reflects back a portion of this spherically expanding gas, then a force will be exerted on the plate equal to twice the ultimate gas velocity times the mass flow rate. If  $n$  bombs of mass  $m$  explode per second, the mass flow rate out from the center of explosion is  $nm$  which would be the fuel consumption rate. If the plate subtends an angle  $2\Theta$  from the explosion center, then the mass flow rate hitting the plate will be approximately  $\frac{n m \sin^2 \Theta}{4}$

Hence, the thrust is

$$F \approx \frac{2 n m \sin^2 \Theta}{4} \times 1,800,000$$

yielding a specific impulse of:

$$I = \frac{F}{n m g} \approx 28,000 \sin^2 \Theta \quad (22)$$

If the space ship subtends a half angle of  $20^\circ$  to the bomb explosion center, then  $\sin^2 \Theta = .115$  giving a specific impulse of about 3200 seconds using these very arbitrary assumptions. So you see, the idea is plausible. Because the project is classified, I am reluctant to say more about Project ORION.

## VII. THERMONUCLEAR ROCKETS

In order to release thermonuclear energy, a plasma of light elements must be heated to a temperature of one billion degrees Kelvin. At such temperatures, a portion of the ions are moving at sufficient speeds to cause fusion upon collision, accompanied by the release of large energies.

Four frequently considered fusion reactions are shown on Slide 41. The amount of energy liberated to each particle is given in million-electron-volt units. One electron volt is equivalent to 11,605°K. The first two reactions occur with equal probability and are between deuterium ions. The third and fourth are between deuterium and either tritium or helium 3 respectively. The difficulty with using deuterium-deuterium and deuterium-tritium reactions is that a large fraction of the energy appears as high velocity neutrons.

At the temperature ranges of interest, only magnetic fields offer promise as a means of confinement. The neutrons are unaffected by magnetic field and are thus lost from the reaction zone. Recovery of this energy in a cooled shield would only complicate a thermonuclear space propulsion system. Hence, reactions liberating charged particles that can be trapped by magnetic fields are preferred.

Deuterium and helium 3 might be provided as the fuel utilizing the fourth reaction. If the reactor temperature is held at a sufficiently high value, the probability of a deuterium helium 3 reaction is much greater than the deuterium-deuterium reaction so that only about 5% of the energy would be liberated as neutrons.

The reacting plasma would be contained in a magnetic bottle as shown in Slide 42. The charged particles are reflected back toward the reactor interior by the strong fields on the ends. The plasma pressures of more than 1000 pounds per square inch suggest confining field strengths of over 100 kilogauss. These fields would be provided by superconducting magnets to minimize the power losses associated with containment. The field on one end of the reactor would be weaker than on the other end, which would allow propellant to flow through the magnetic nozzle to space.

The cryogenic magnet must, of course, be cooled to low temperatures with a liquid helium system. To minimize the heat load on the magnet due to bremsstrahlung and neutron radiation, shields are provided as shown on Slide 43. The thermal capacity of the hydrogen cools the cryopant and the neutron or "secondary" shield. This hydrogen is ejected by the reactor exit jet. Additional cooling through a radiator system is required for the bremsstrahlung or "primary" shield.

The performance of such a thermonuclear rocket is pretty spectacular. Thrust to engine weight ratios of as high as 0.01 are feasible and correspond to about 1 or 2 kilowatts of jet power per pound of engine weight. The specific impulse would be on the order of 10,000 seconds. The performance of such a system would therefore be about an order of magnitude better than that predicted for the nuclear fission electric propulsion system. Controlled fusion, however, has not yet been obtained in a laboratory reactor. Hence, the thermonuclear rocket won't be a reality for a long time in the future.

### VIII. THE PHOTON ROCKET

The maximum possible value of specific impulse,  $3 \times 10^7$  is obtained in the photon rocket. In this case, however, the power requirements are so high that no known energy source or conversion method is sufficient. Nearly two million horsepower or 1330 megawatts of power would be required for each pound of thrust. Even if such energy sources were available, directing the photons in the jet would require materials with almost perfect reflection coefficients to keep them from overheating.

The photon sail, on the other hand, might be practical for some space missions. The maximum thrust on a photon sail at the earth's distance from the sun is about  $1.96 \times 10^{-7}$  pounds per square foot.

If the sail were oriented (Slide 44) to give maximum thrust tangential to the path, then thin plastic reflectors might yield tangential thrust-weight ratio of about  $2 \times 10^{-5}$  (assuming a plastic thickness of 0.0005 in.). The solar sail might therefore be an interesting propulsion system for instrumented space probes. It can sail either toward or away from the sun simply by controlling the direction of the tangential thrust component.

The greatest effectiveness of the solar sail would be for flights near the sun. At the Venus orbital distance from the sun, the tangential thrust to weight ratio has increased to  $3.86 \times 10^{-5}$ ; at Mercury the value is  $1.32 \times 10^{-4}$ .

Solar sails might even be useful in the earth-satellite space region. A plastic disk, for example, might be spin stabilized to have an orientation  $45^\circ$  to the sun's rays (Slide 45). The photon thrust will then always remain in the same direction. As the sail orbits through the earth's shadow, the thrust disappears. Thus the solar sail satellite can change its orbital path through the thrust received on the sunny side.

If one were to get enthusiastic about solar sailing, he would certainly need more information on the properties of thin plastic sheet in the radiation and high vacuum environment of space.

## IX. RADIOISOTOPE SAIL

The radioisotope sail (Slide 46) is perhaps useful for instrumented probes of deep space. An  $\alpha$  emitter would be painted on one surface of a plastic membrane. The emitted  $\alpha$  particles constitute the jet that propels this device. Ideal thrust to weight ratios of  $10^{-4}$  might be obtainable. Precautions would have to be taken, however, to neutralize the charge that would accumulate if electrons did not follow the  $\alpha$  particles into space. Guidance might also be a problem.

A radioisotope photon rocket is also feasible. In this arrangement, a radioisotope powered very hot tungsten capsule would be placed at the focal point of a thin plastic parabolic reflector. The thrust thus generated by the photon beam might propel a small payload to accelerations of perhaps  $10^{-5}$  g.

## X. ANTIGRAVITY PROPULSION

Some people have talked glibly of antigravity as a solution to space propulsion problems in the fond hope that some genius will discover the technique to accomplish this dreamed of breakthrough. If gravity could be cancelled, then an antigravity wave would presumably travel outward from the spaceship to cancel the gravitational attraction that now exists. If this hypothesized antigravity wave travelled with the speed of light, then the power requirements for propulsion would be identical to those of the photon rocket - 1330 megawatts per pound of gravity cancellation. Thus an antigravity propulsion device would be impractical even if it were possible.

CG272-17

## OPTIMIZATION OF SPACE PROPULSION SYSTEMS

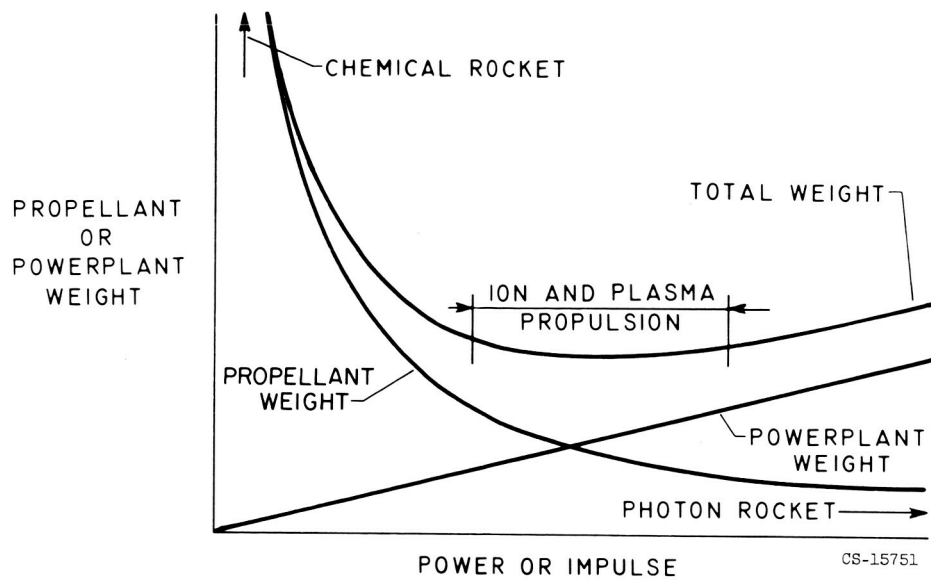


Fig. 1.

## SOLID CORE HEAT TRANSFER TYPE NUCLEAR ROCKET

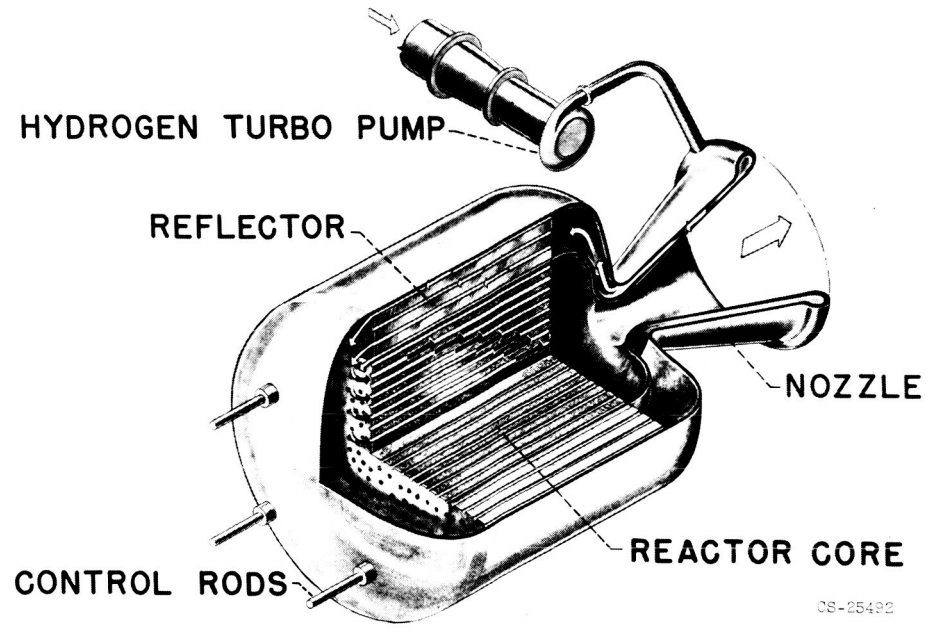
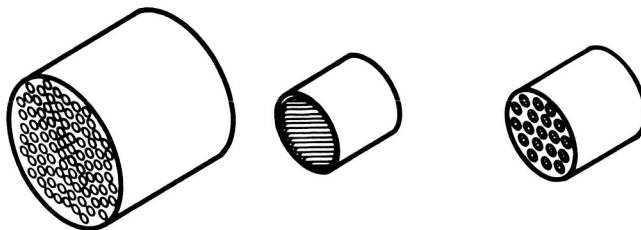


Fig. 2.

## NUCLEAR ROCKET REACTORS

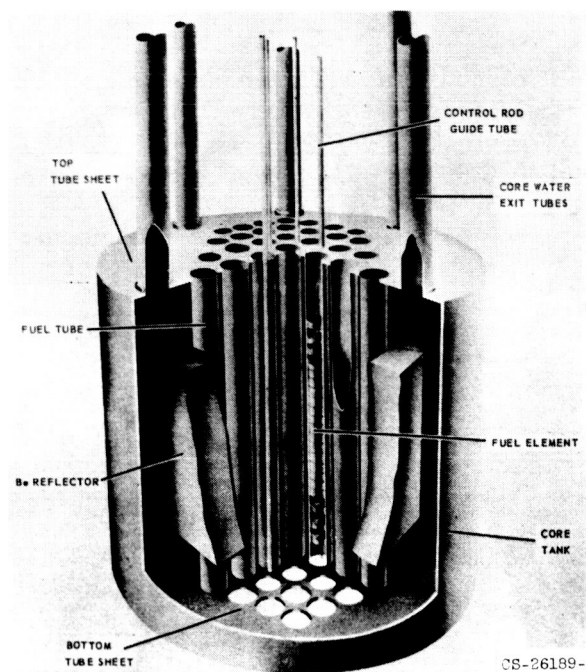


REACTOR TYPE	HOMOGENEOUS THERMAL (HOT MODERATOR)	FAST (NO MODERATOR)	HETEROGENEOUS THERMAL (COOLED MODERATOR)
MODERATING MATERIAL	GRAPHITE BERYLLIUM OXIDE	NONE	WATER HEAVY WATER BERYLLIUM BERYLLIUM OXIDE METALLIC HYDRIDES
FUEL BEARING MATERIAL	COATED GRAPHITE BERYLLIUM OXIDE	REFRACTORY METALS (W,Mo) CARBIDES (ZrC,HfC,TaC)	REFRACTORY METALS ( $W^{184},Mo$ ) COATED GRAPHITE CARBIDES (ZrC)

CS-25486

Fig. 3.

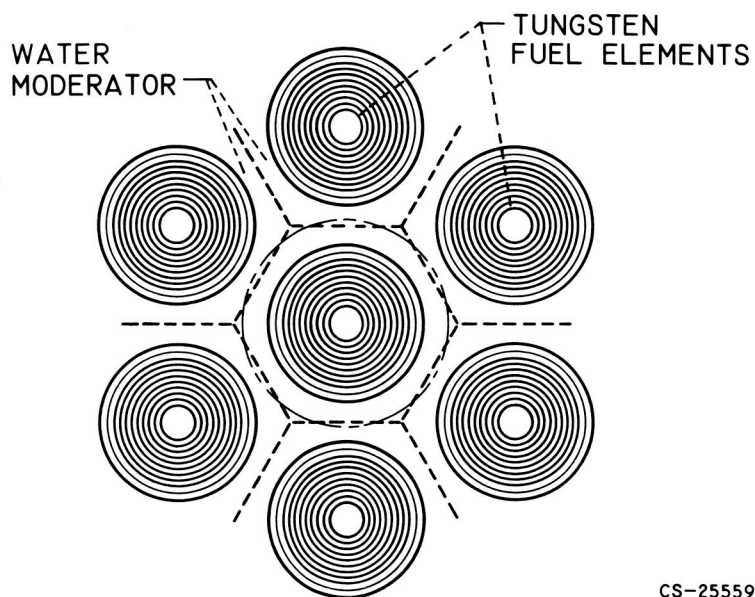
## HTRE NO.1 CORE



CS-26189-A

Fig. 4.

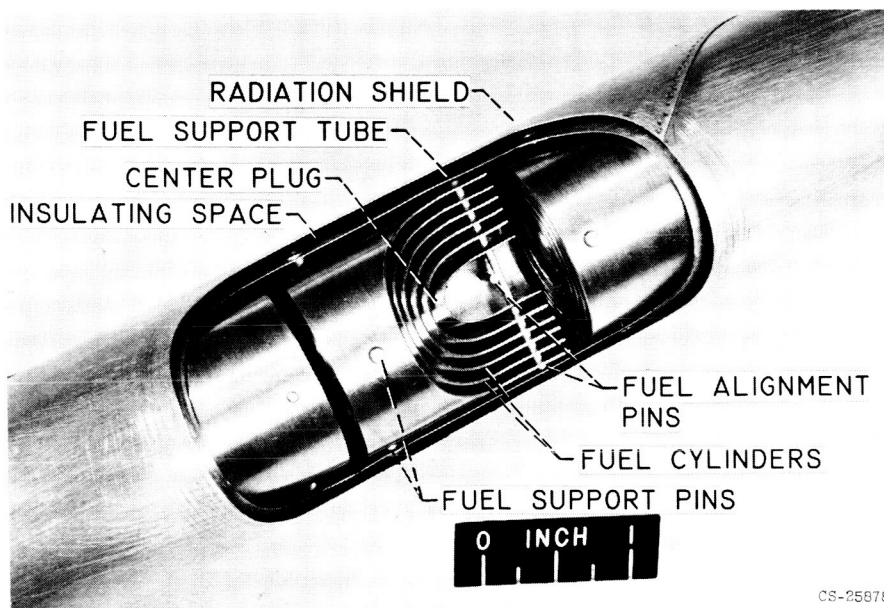
# TYPICAL FUEL ELEMENT ARRAY FOR HETEROGENEOUS CORE



CS-25559

Fig. 5.

## FUEL ELEMENT ASSEMBLY W-H<sub>2</sub>O REACTOR CONCEPT W-H<sub>2</sub>O REACTOR CONCEPT



CS-25878

Fig. 6.

## WATER MODERATED REACTOR

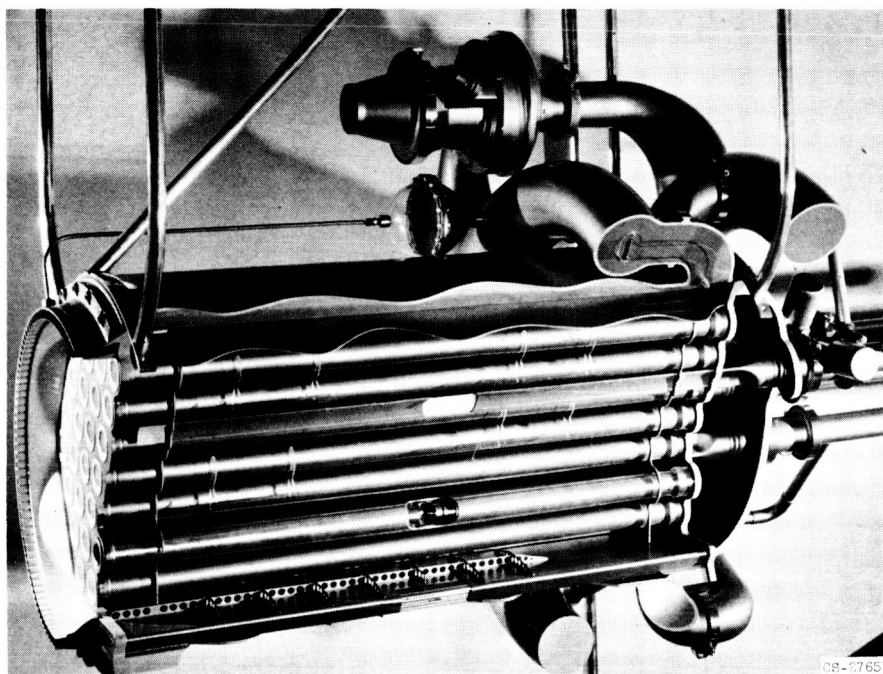


Fig. 7.

## TUNGSTEN TOTAL CROSS SECTION

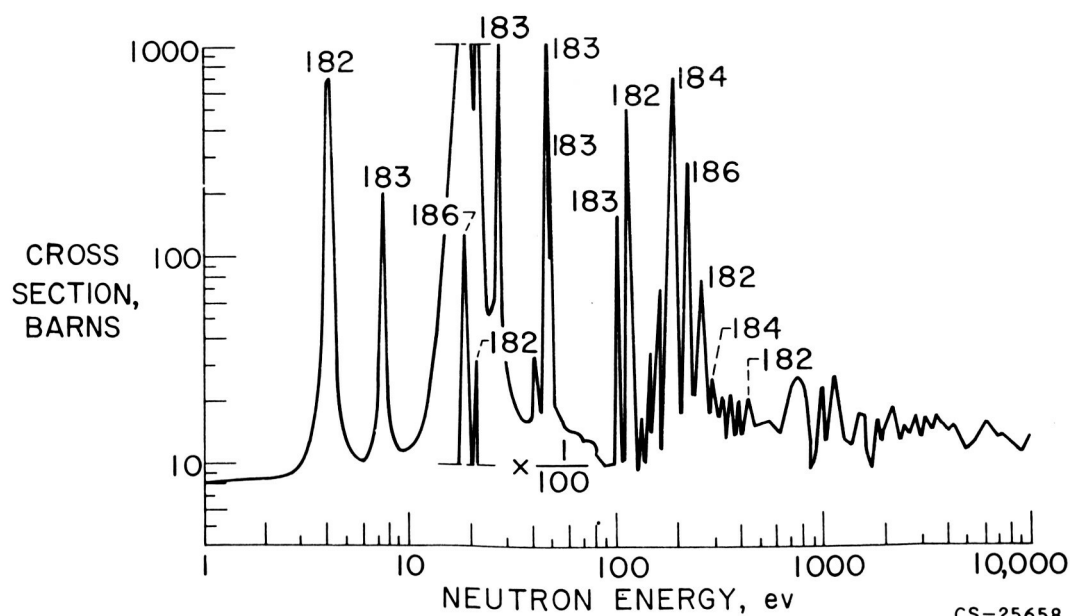
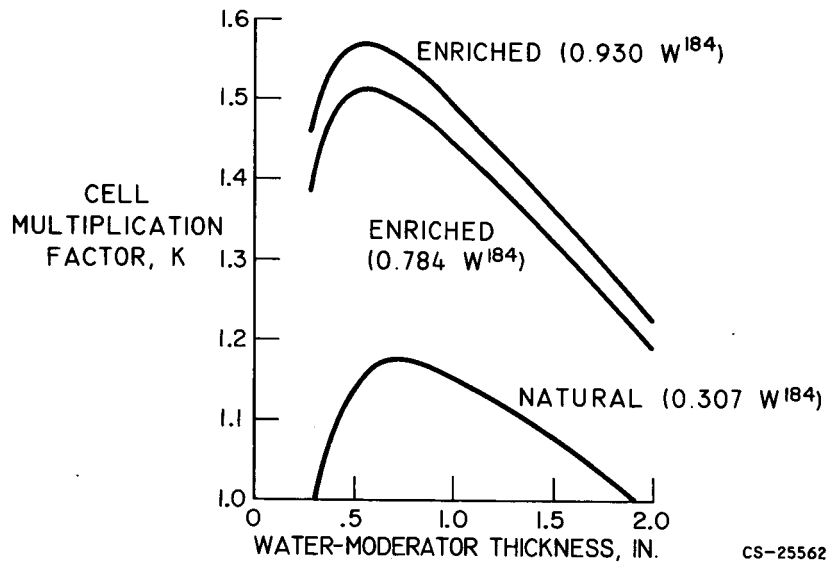


Fig. 8.



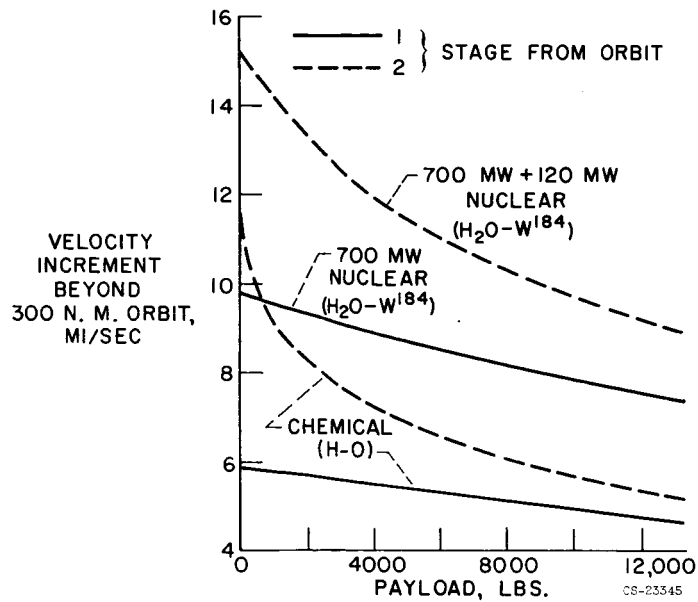
# MULTIPLICATION FACTOR FOR TUNGSTEN FUEL CELL TUNGSTEN SLABS 0.10 IN. THICK IN WATER



CS-25562

Fig. 9.

# NUCLEAR ROCKET PROBE PERFORMANCE BOOSTER, SATURN C-4 INITIAL WEIGHT IN ORBIT, 150,000 LBS



CS-23345

Fig. 10.

# CONSTANT-THRUST TRAJECTORY FROM SATELLITE ORBIT

THRUST/WEIGHT =  $10^{-4}$

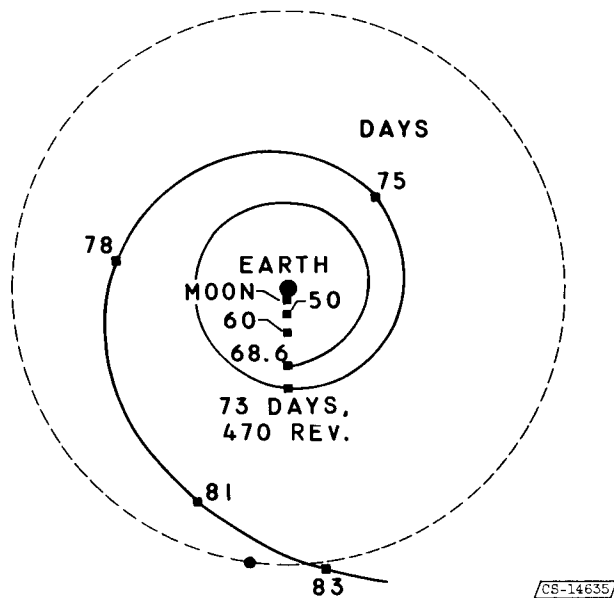


Fig. 11.

## PERFORMANCE OF ELECTRIC ROCKET WITH CONSTANT PROPULSION TIME, $t$

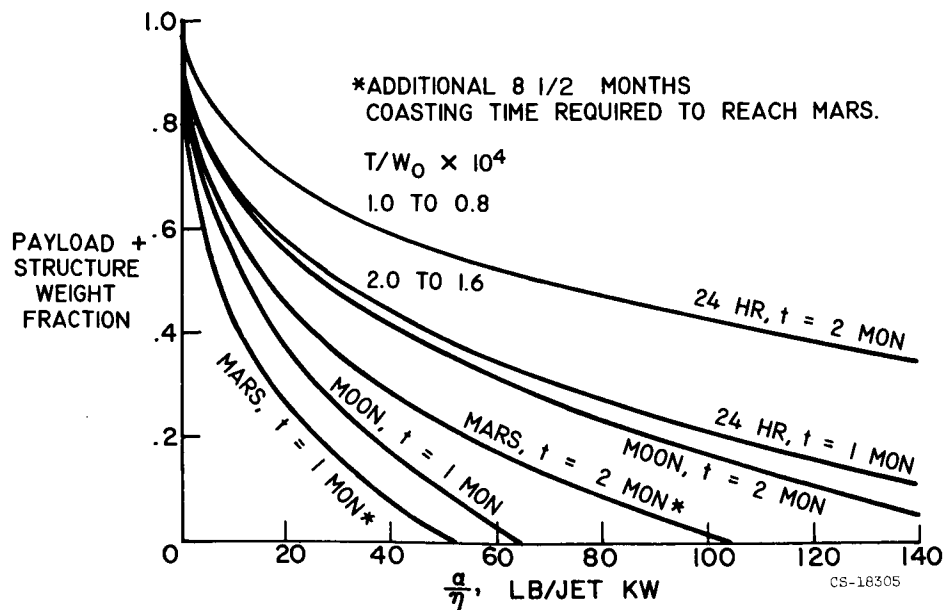
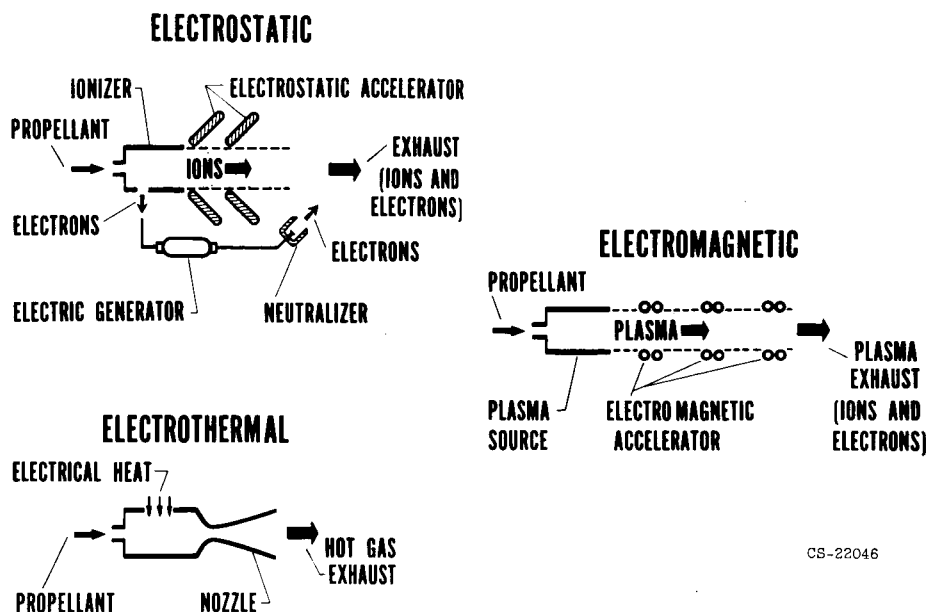


Fig. 12.

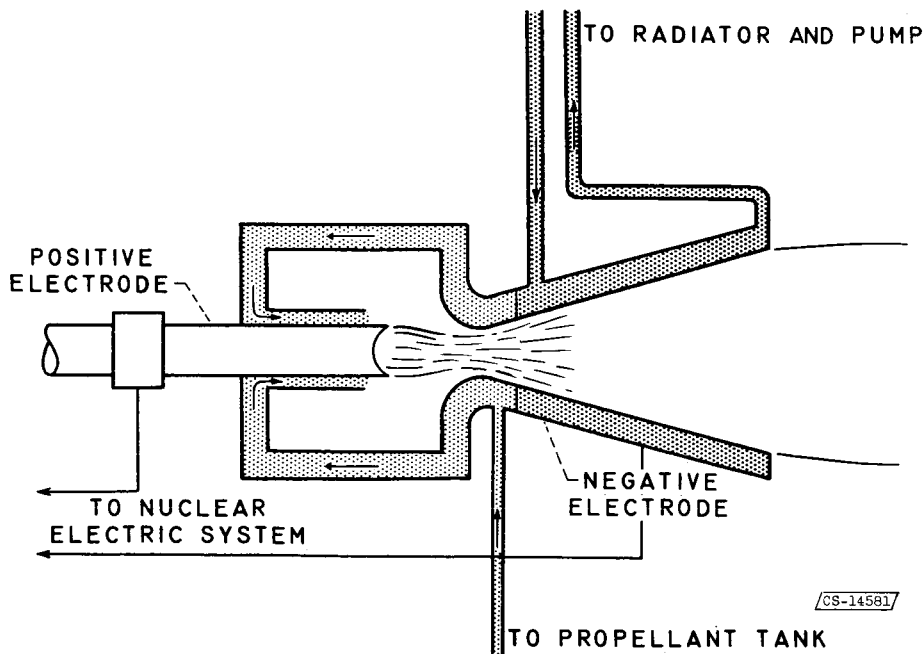
# GENERAL CLASSIFICATION OF ELECTRIC ROCKET THRUSTORS



CS-22046

Fig. 13.

## ARC-JET PROPULSION SYSTEM



CS-14581

Fig. 14.

## LEWIS RESISTO-JET

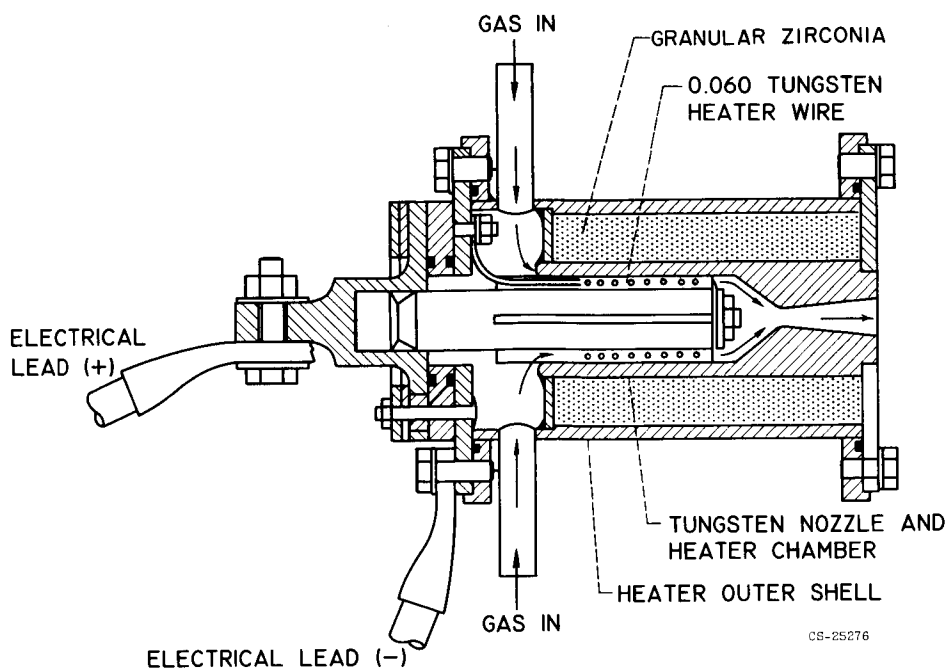


Fig. 15.

## CONTACT ION ENGINE SCHEMATIC

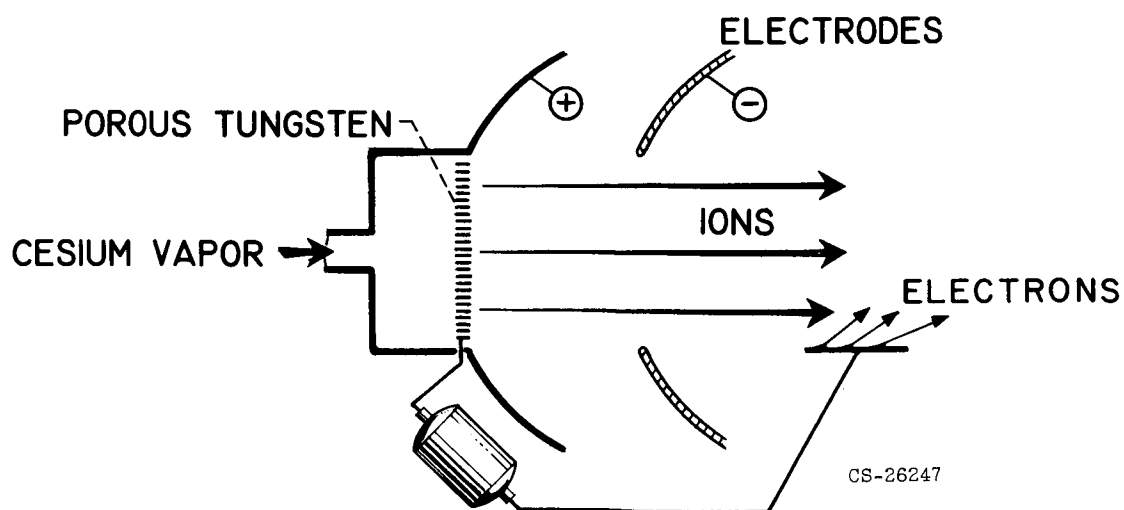


Fig. 16.

# ION BEAM PHOTO

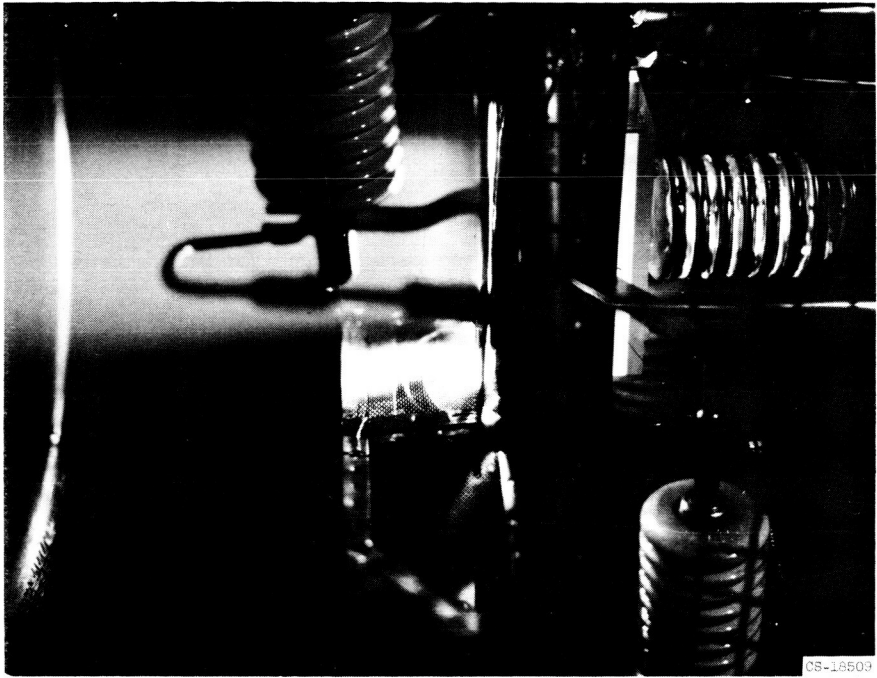
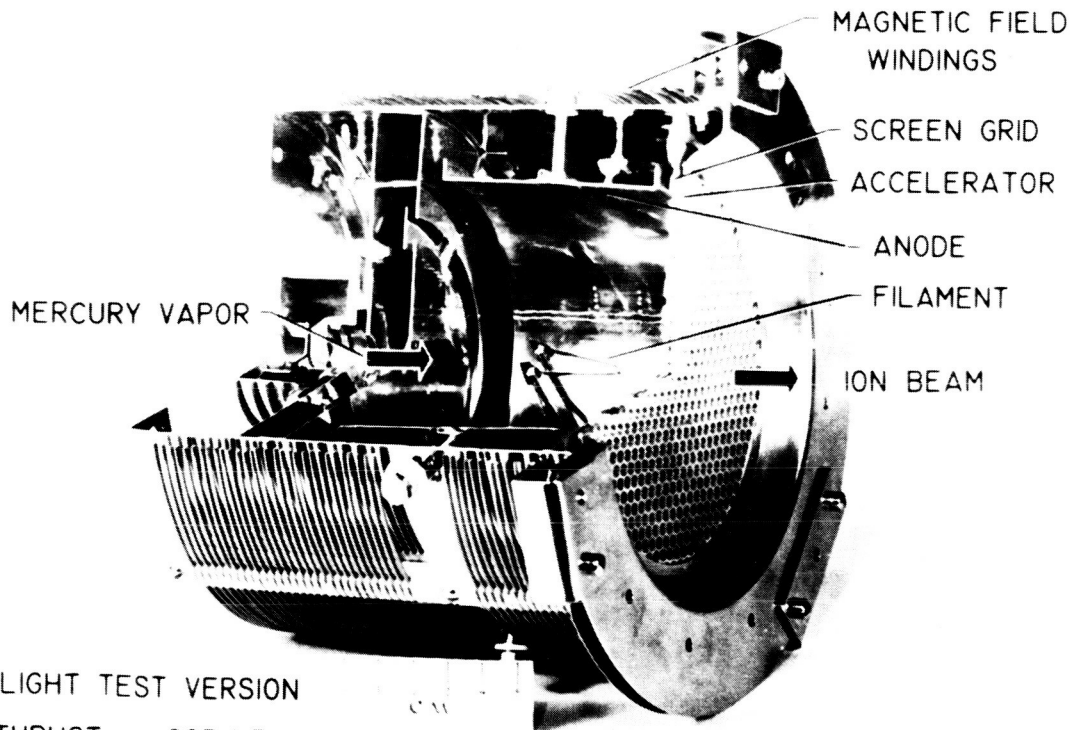


Fig. 17.

## LEWIS ELECTRON-BOMBARDMENT THRUSTOR



FLIGHT TEST VERSION  
THRUST = .007 LB  
SPECIFIC IMPULSE = 5000 SEC

CS-23047

Fig. 18.

## ELECTROSTATIC THRUSTOR EFFICIENCIES

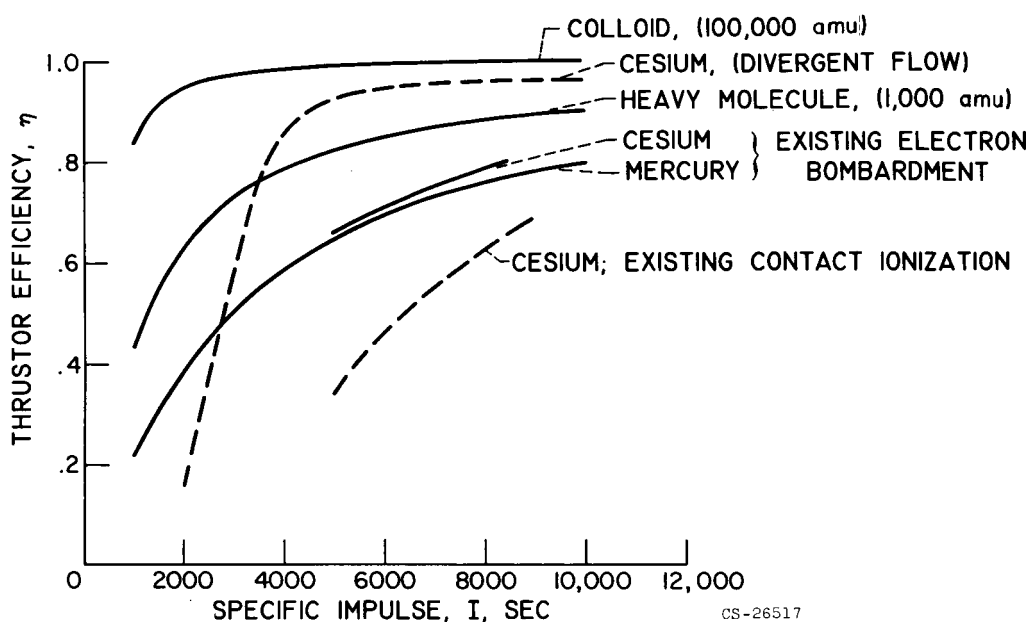


Fig. 19.

## COLLOIDAL-PARTICLE THRUSTOR

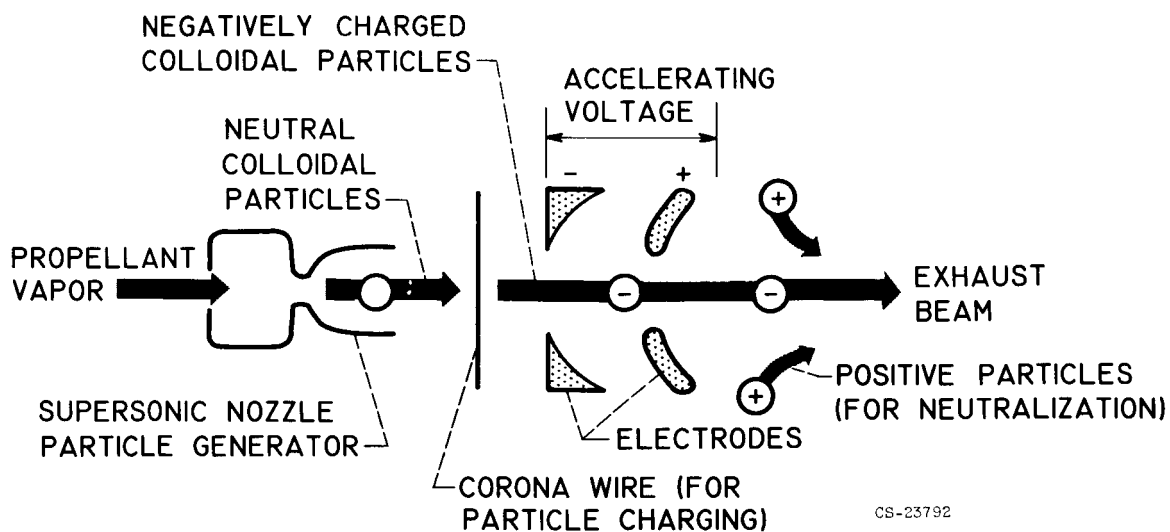


Fig. 20.

NASA LEWIS EXPERIMENTAL  
COLLOIDAL-PARTICLE THRUSTOR  
CORONA-DISCHARGE CHARGING

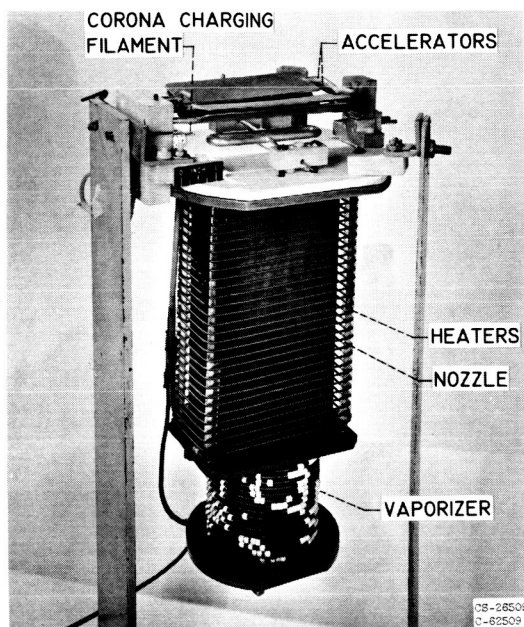


Fig. 21.

NASA LEWIS EXPERIMENTAL  
COLLOIDAL-PARTICLE THRUSTOR  
ELECTRON-BOMBARDMENT CHARGING CHAMBER

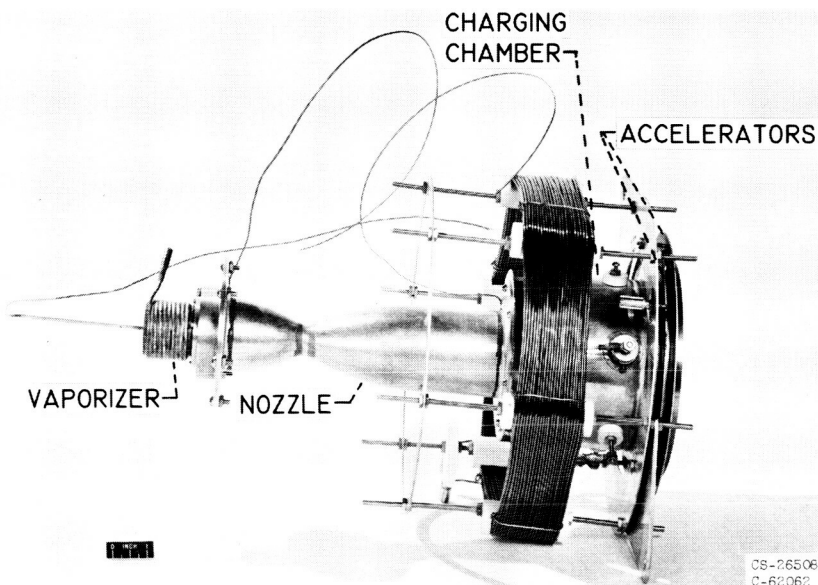


Fig. 22.

# VON ARDENNE ION ROCKET

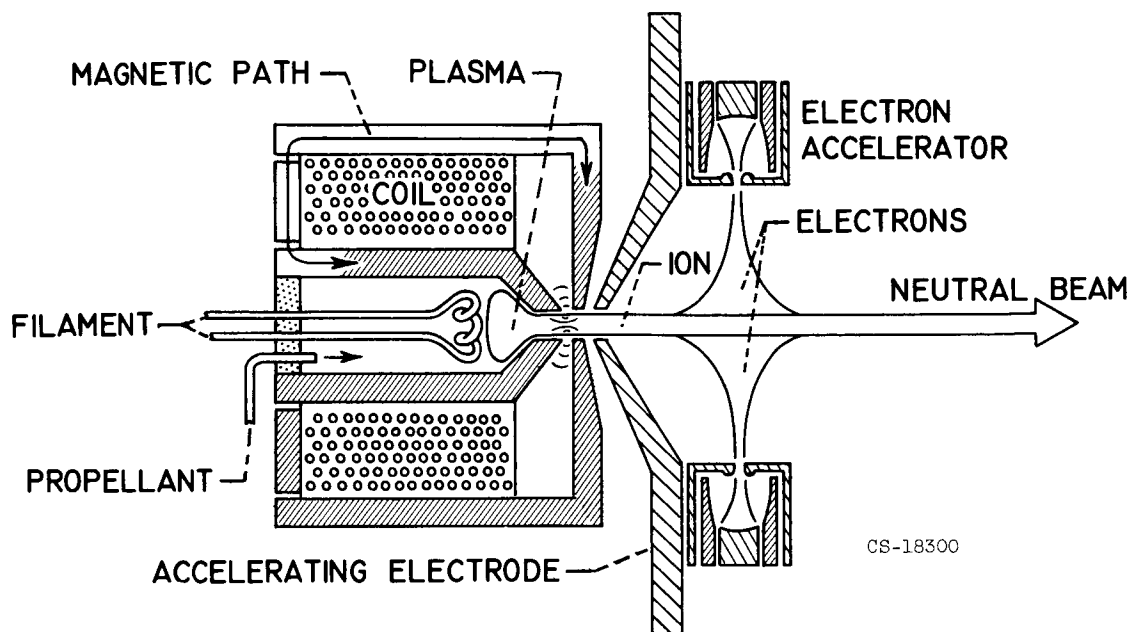


Fig. 23.

# HALL CURRENT ION ACCELERATOR

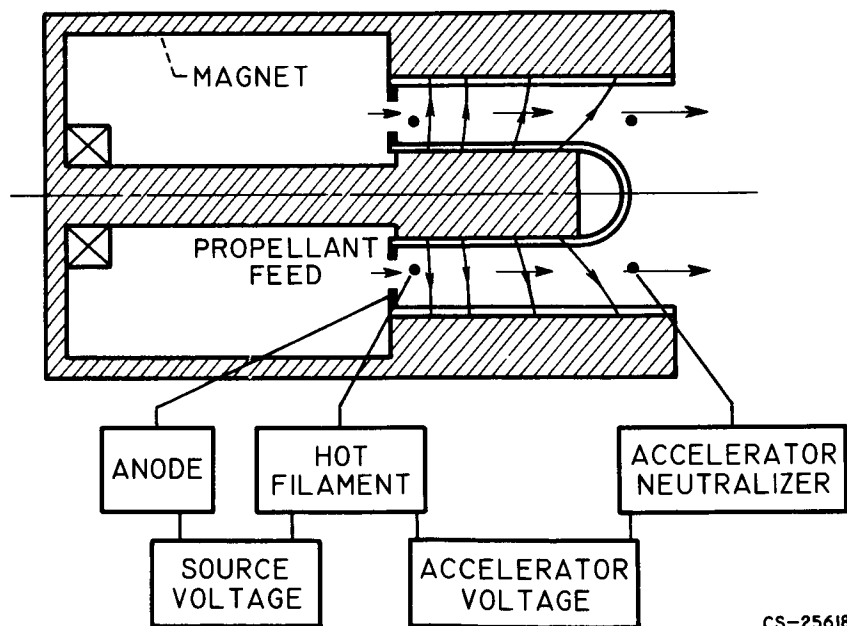


Fig. 24.



# CHARGED PARTICLE MOTIONS IN MAGNETIC AND ELECTRIC FIELDS

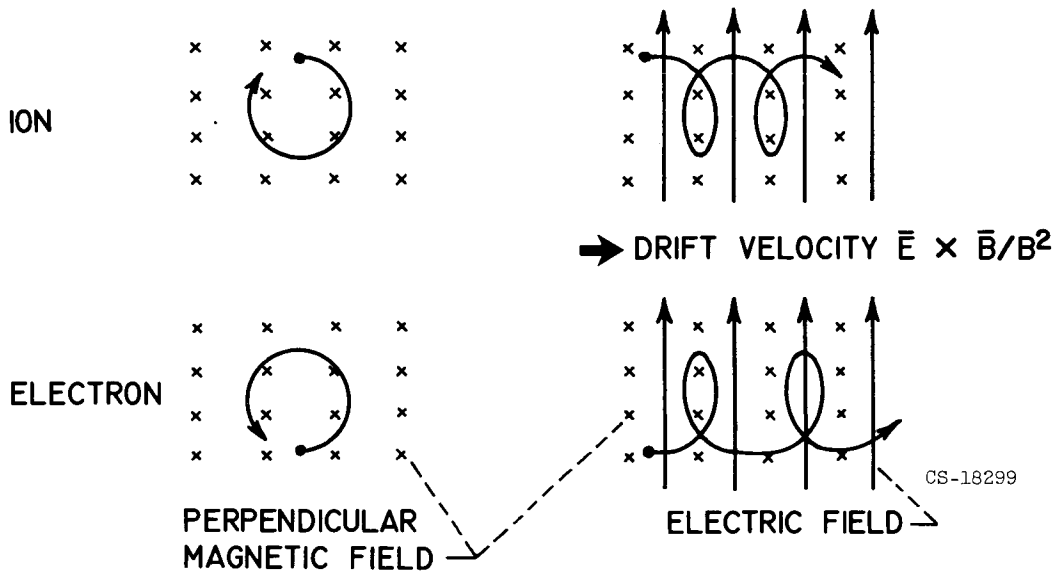


Fig. 25.

## E X B ACCELERATOR

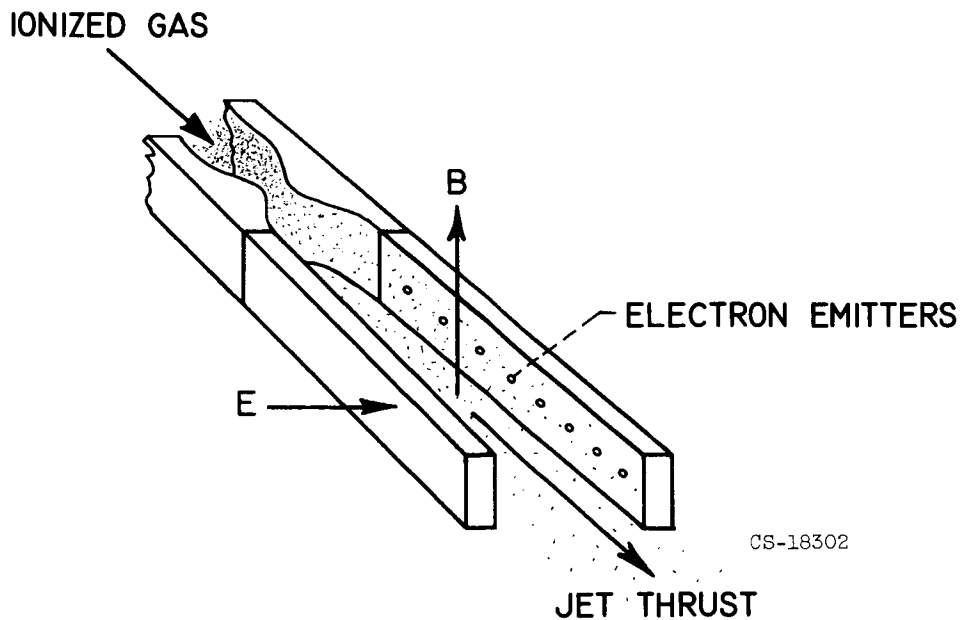
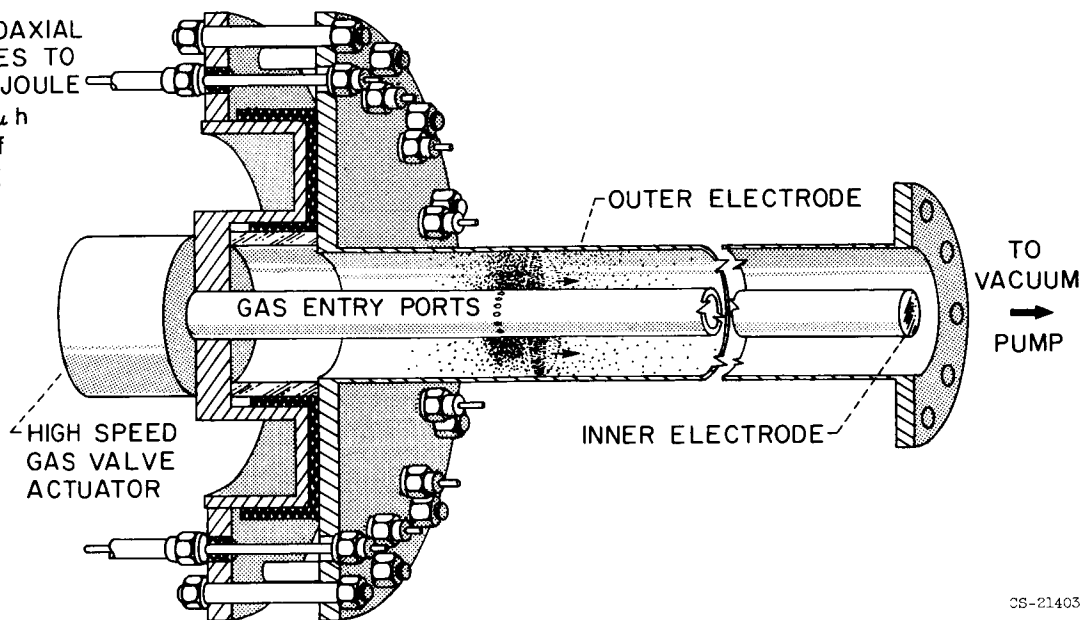


Fig. 26.

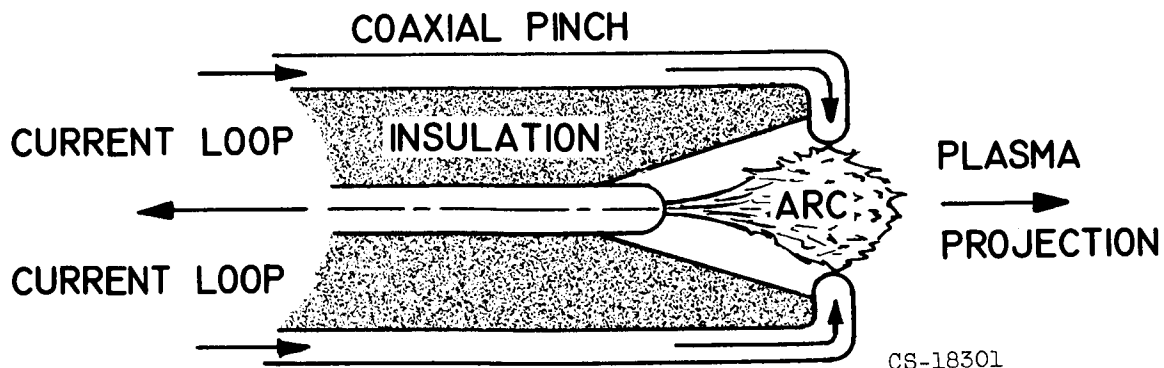
33 COAXIAL  
CABLES TO  
5450 JOULE  
10 m  $\mu$ h  
12.1  $\mu$ f  
BANK



CS-21403

Fig. 27.

## PLASMA PROJECTION VIA THE PINCH EFFECT



CS-18301

Fig. 28.

# PLASMA ROCKET

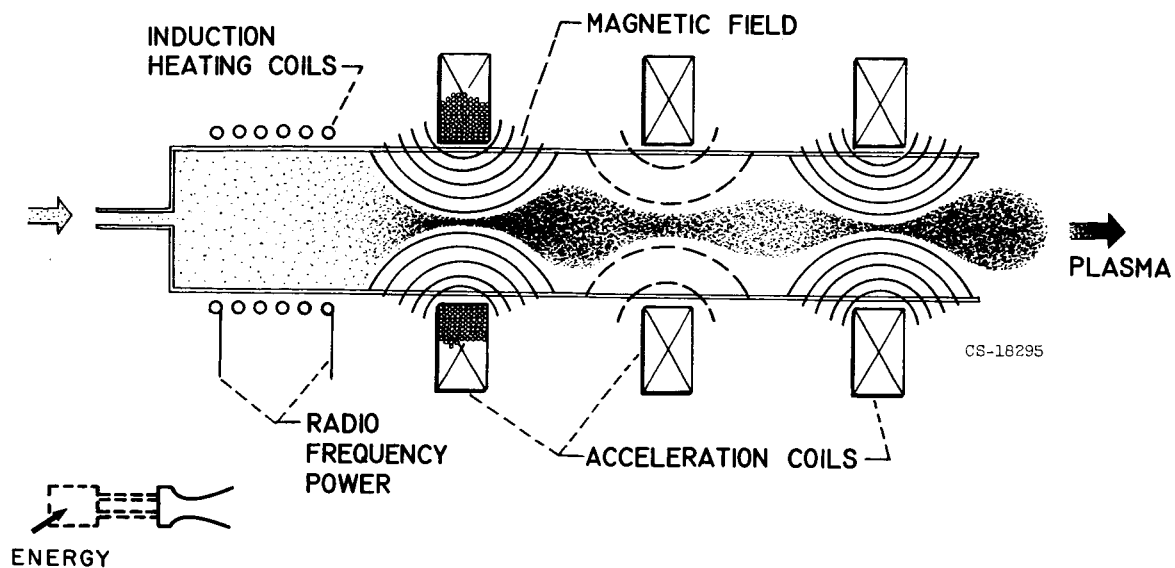


Fig. 29.

## MANNED MARS MISSION

CREW SHIELDING FOR 100 REM DOSE, METEOROID SHIELDING FOR  $P_0$   
 $= 0.999$

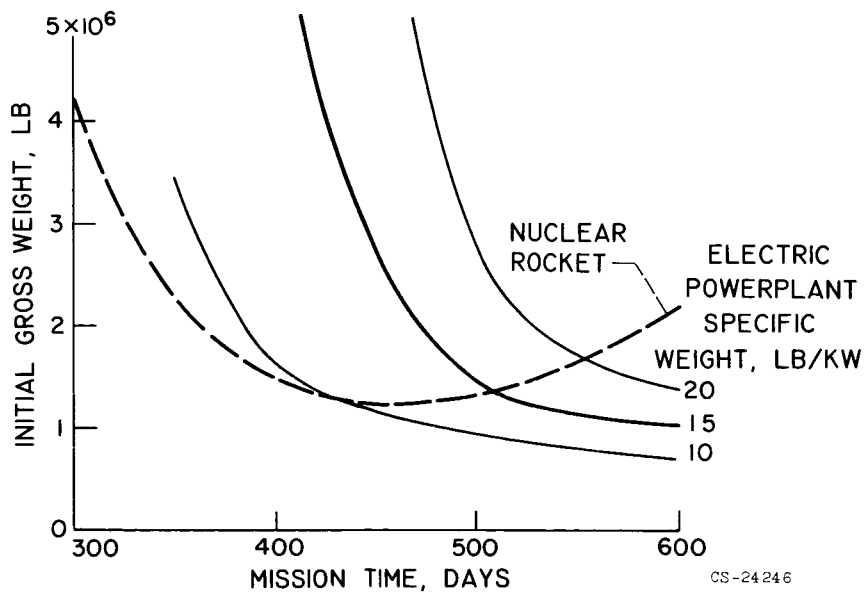


Fig. 30.

# ELECTRIC ROCKET AND NUCLEAR ROCKET EQUAL PAYLOAD WEIGHT

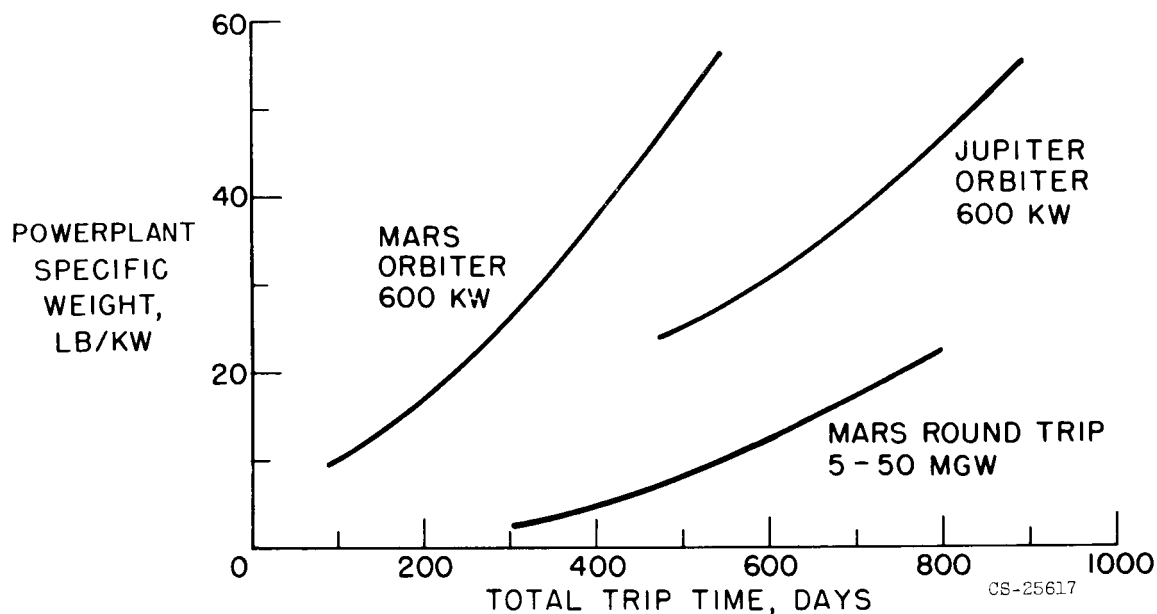


Fig. 31.

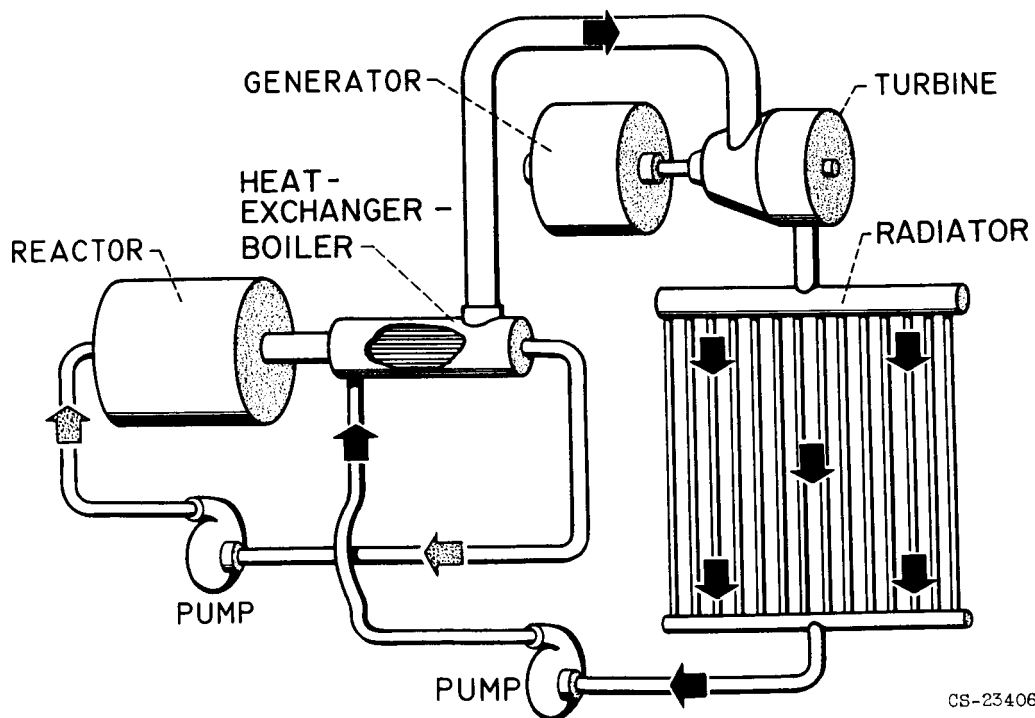


Fig. 32.

# SCHEMATIC OF BRAYTON CYCLE SPACE POWER SYSTEM

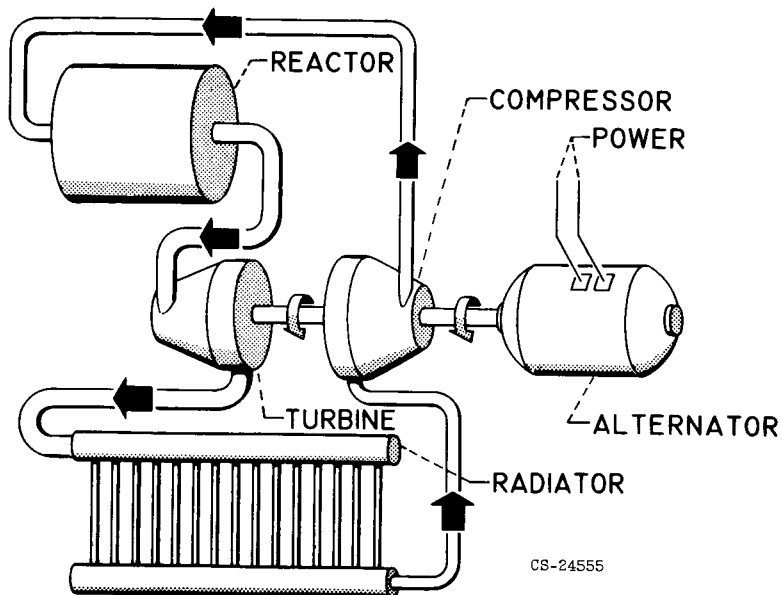


Fig. 33.

# THERMIONIC CONVERTER

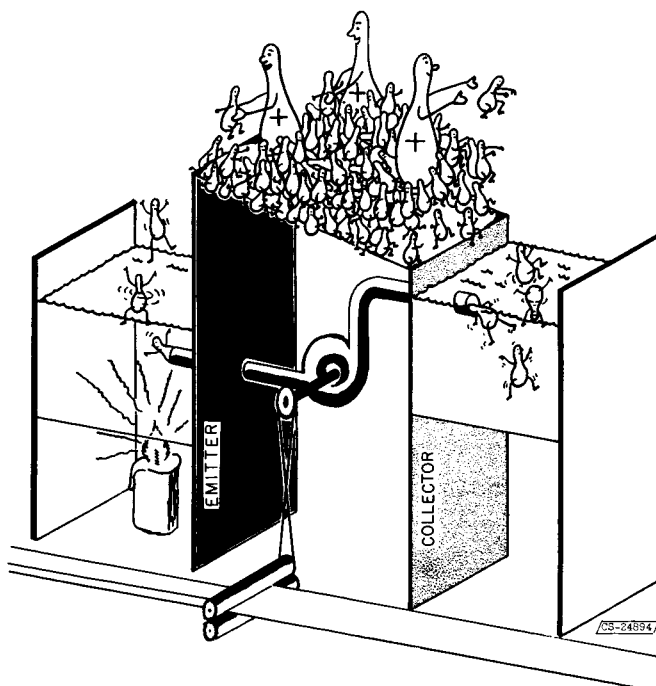


Fig. 34.

## NUCLEAR THERMIONIC POWER SYSTEM

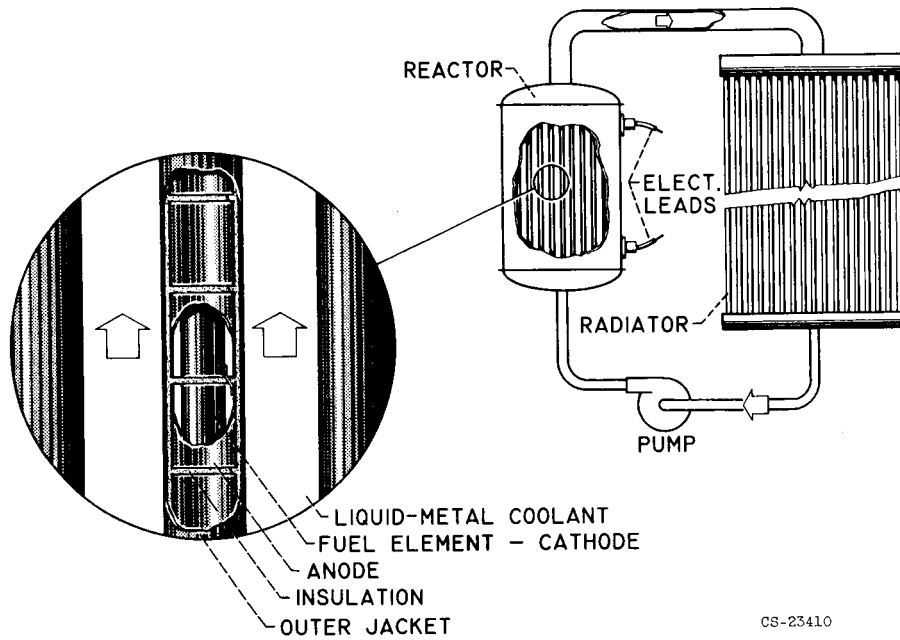


Fig. 35.

## GASEOUS CAVITY REACTOR

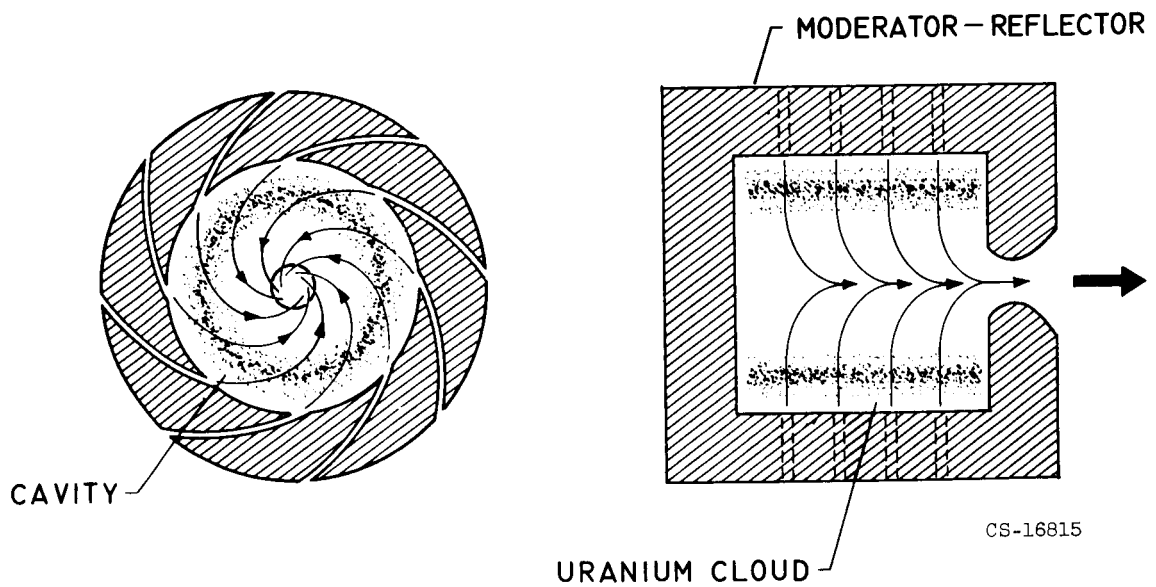


Fig. 36.

## VORTEX GAS CORE REACTOR

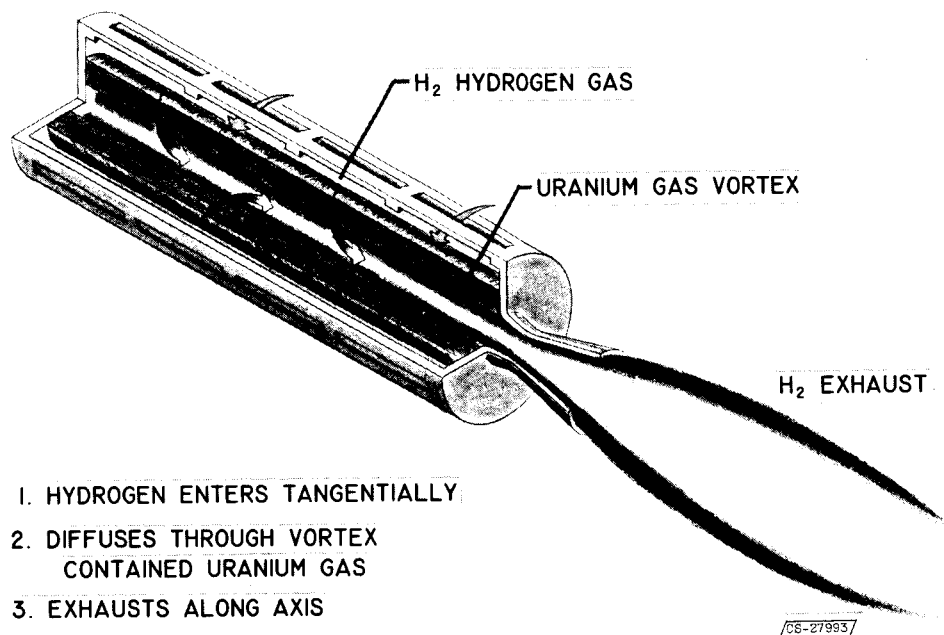


Fig. 37.

## VORTEX TUBE AND VORTEX MATRICES

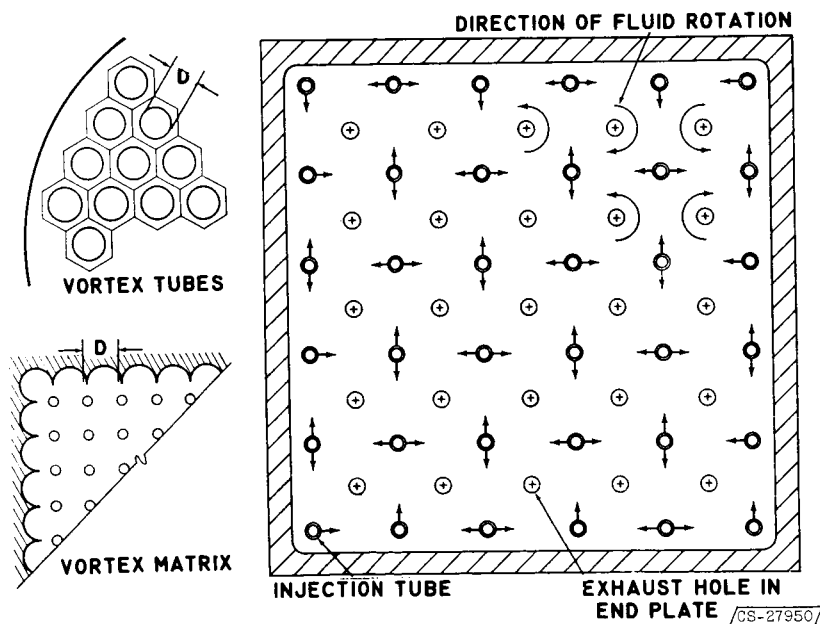
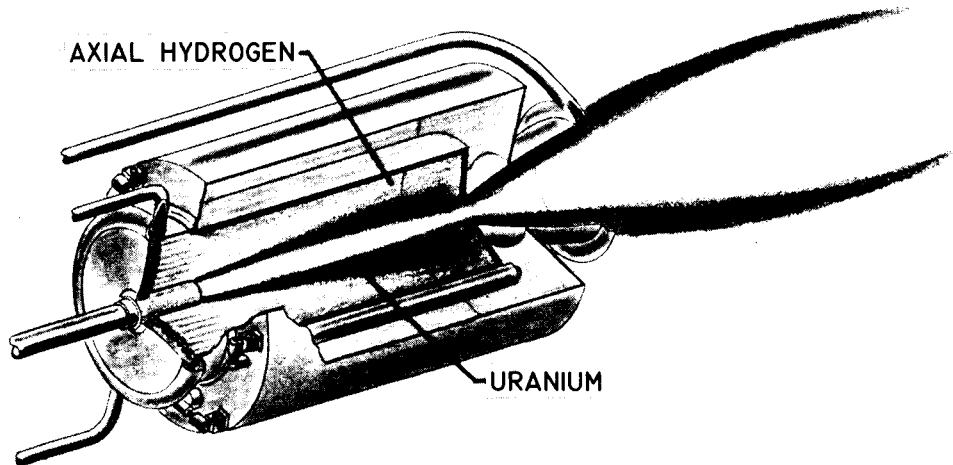


Fig. 38.

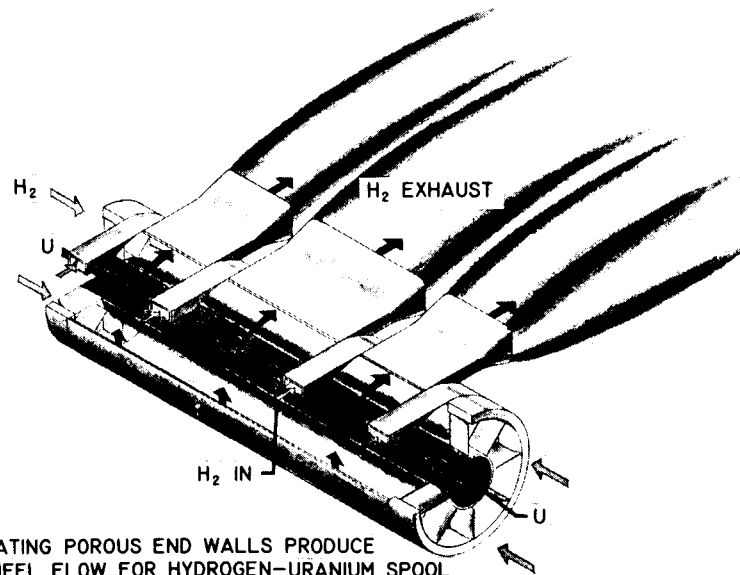
## COAXIAL JET REACTOR



1. URANIUM AND HYDROGEN FLOW AXIALLY
2. URANIUM FLOWS MUCH SLOWER THAN HYDROGEN
3. TURBULENT MIXING GRADUALLY CONSUMES URANIUM CS-27997
4. LOW SPEED HYDROGEN BUFFER LAYER MINIMIZES URANIUM LOSS

Fig. 39.

## WHEEL FLOW REACTOR

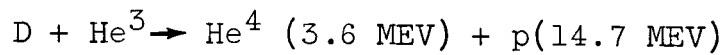
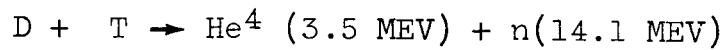
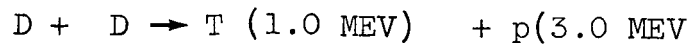
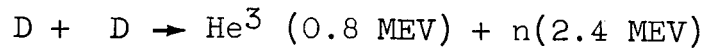


1. ROTATING POROUS END WALLS PRODUCE WHEEL FLOW FOR HYDROGEN-URANIUM SPOOL
2. AXIAL VELOCITIES TAILORED TO GIVE MINIMUM SHEAR
3. PRINCIPAL HYDROGEN FLOW ENTERS AND LEAVES TANGENTIALLY CS-27995

Fig. 40.



## FUSION REACTIONS



CS-23948

Fig. 41.

## THERMONUCLEAR ROCKET

### BASIC COMPONENTS

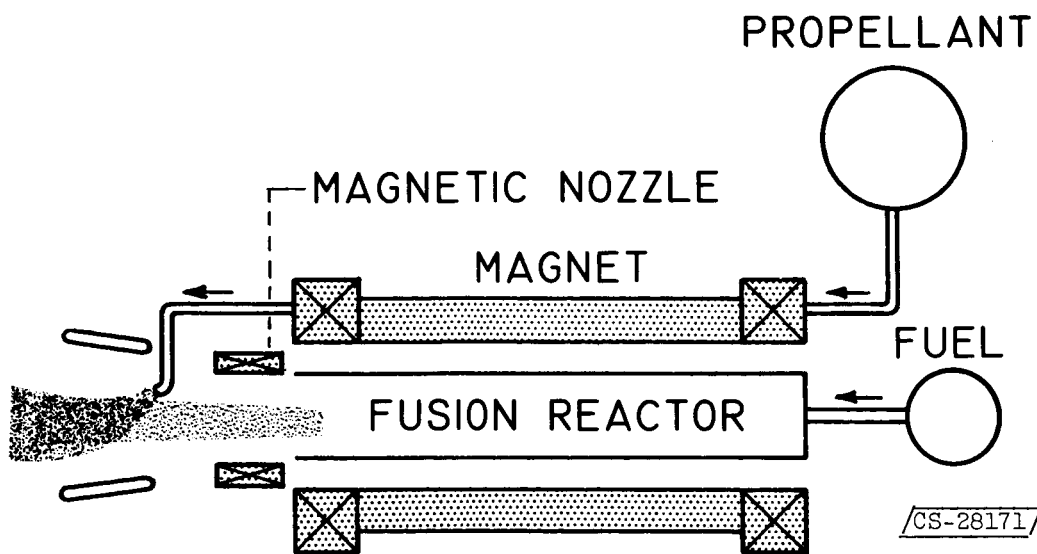


Fig. 42.

## THERMONUCLEAR ROCKET

INCORPORATION OF SHIELDED AND  
CRYOGENICALLY COOLED MAGNET

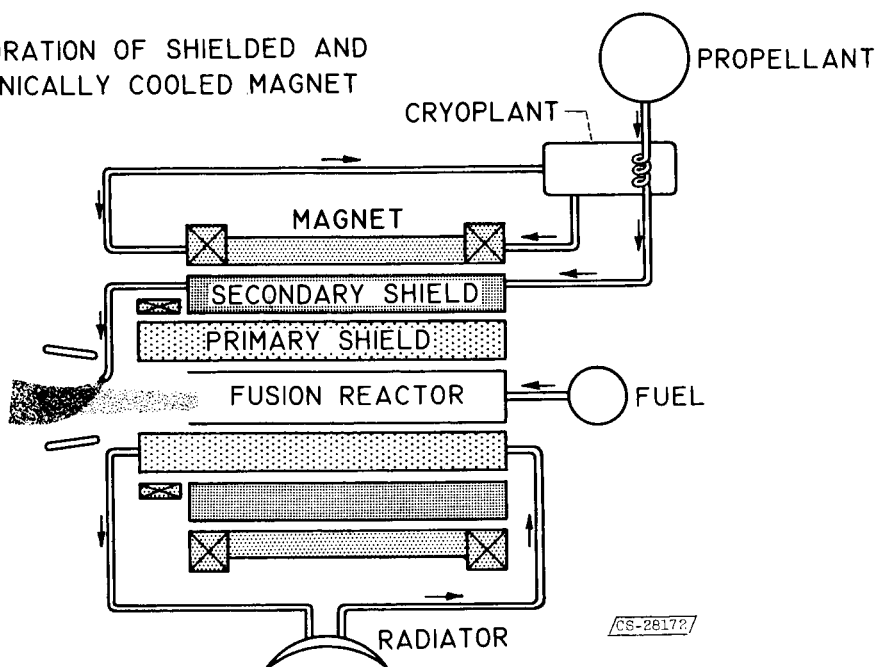
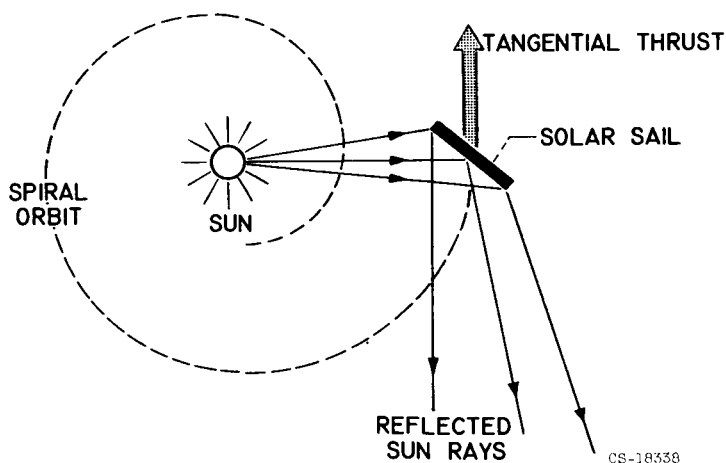


Fig. 43.

## THE SOLAR SAIL



POSITION	MAX THRUST	TANGENTIAL THRUST	TANGENTIAL THRUST TO WEIGHT RATIO
EARTH	$1.96 \times 10^{-7}$ LB/SQ FT	$.755 \times 10^{-7}$	$2. \times 10^{-5}$
VENUS	$3.79 \times 10^{-7}$	$1.46 \times 10^{-7}$	$3.86 \times 10^{-5}$
MERCURY	$12.9 \times 10^{-7}$	$4.97 \times 10^{-7}$	$13.5 \times 10^{-5}$

Fig. 44.

# THE SOLAR SAIL IN EARTH ORBIT

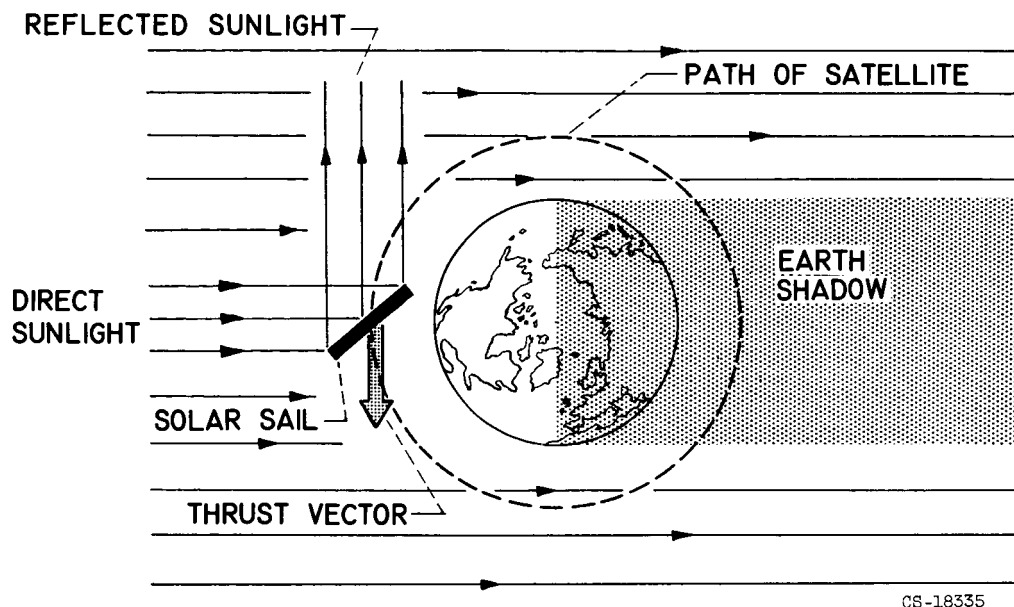


Fig. 45.

## RADIOISOTOPE SAIL

THRUST/SQ FT	$1 \times 10^{-6}$
WEIGHT/SQ FT ( $t = 0.0012''$ )	$9 \times 10^{-3}$
THRUST/WEIGHT (IDEAL)	$1 \times 10^{-4}$

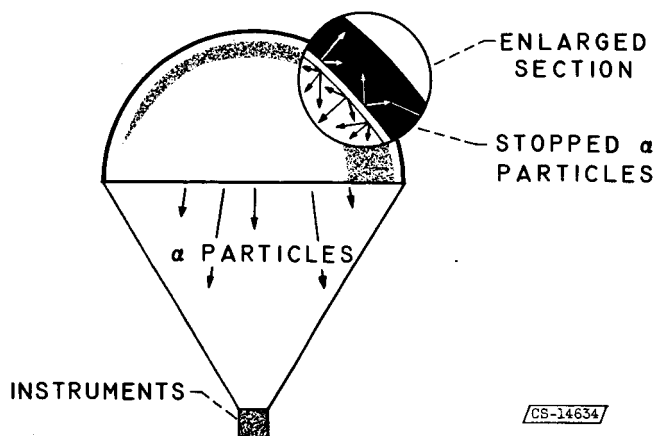


Fig. 46.